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THESIS

**ORBITAL DEBRIS:
COST IMPACT
ON SETTING POLICY**

by

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June, 1996

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ORBITAL DEBRIS: COST IMPACT ON SETTING POLICY

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ABSTRACT

As the exploration of space increases, the problems associated with orbital debris also increase. Orbital debris continues to grow at a linear rate with an ever increasing possibility of a shift to an exponential rate. If this point is achieved, space travel wil,l at best, be extremely hazardous and at worst, unusable. When mitigating orbital debris, cost and policy issues must be addressed. Currently, no policy exists that makes the mitigation of orbital debris mandatory but it only strongly recommends mitigation if it is cost effective. This thesis addresses the cost impact of alternative spacecraft design options for orbital debris mitigation. The cost impact is shown by developing generic satellite characteristics, considering two different altitudes, and using alternative design options.

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I. INTRODUCTION

For as long as humans have gazed into the skies, the desire to unfold the secrets held in the vast darkness of space has been strong and continues to the present day. Within the last forty years, technological advances have allowed the ability to start discovering some of these secrets. During this time, over 3,400 spacecraft (all within the last 40 years) have been launched into the near Earth orbit environment [Ref.1]. This technological explosion has had a side effect - orbital debris. Of the 23,000 objects cataloged as orbital debris, approximately 8000 still remain in orbit. The number of trackable (cataloged) objects in orbit increases by approximately 200 per year [Ref 2]. However, there are many more smaller untrackable objects in orbit. A collision with any of them could cause catastrophic damage and create many more objects in orbit. In the author's opinion, orbital debris will eventually become a major problem if steps are not taken to limit or reduce debris growth in the near future.

Orbital debris issues are analogous to terrestrial environmental issues. For years recycling did not exist. Only recently, when the environment was starting to be threatened, did recycling become commonplace. This was a result of educating the"users of the environment" and setting policy. Orbital debris is no different. It is “trash” that is polluting space and action is needed to ensure the continued safe use of space. Unfortunately terrestrial environmental clean up let alone space clean up does not come

cheap. So when considering orbital debris reduction measures, cost impact must be considered. In fact the very limited orbital debris policies that are currently in effect urge users of space to minimize orbital debris generation as long as it is "cost effective" [Ref. 3].

The exact mix of extra cost and policy setting to achieve orbital debris reduction is difficult to determine. This thesis will first develop a base case of satellite characteristics. Next, it will study three critical parameters regarding designing and building a satellite and show how they impact lifecycle costs. These parameters are fuel, altitude, and decay lifetime. This thesis has five major components: background, lifecycle cost model, option analysis, sensitivity analysis, and summary and conclusions.

Within the background section, definitions of debris, types of debris, and previous policy regarding orbital debris are covered. This section also provides a background summary on orbital debris issues.

The second major section develops the base case, and uses two different altitudes to analyize three options for orbital debris mitigation. Lifecycle cost and decay time of each option is compared showing their benefits and trade-offs. The different options are (1) After full completion of its operational period, the satellite would be left on-orbit to decay naturally; (2) The satellite would be deorbited prior to full completion of its operational period in order to successfully decay the satellite within NASA's guidelines; (3) The satellite is "redesigned" with a larger fuel load to allow for full operational mission completion and a successful deorbit within NASA's guidelines.

The third section addresses the critical parameters with respect to designing, building, and operating a satellite that impact orbital debris mitigation. The critical parameters are fuel, altitude, and decay lifetime. The results of adjusting the critical parameters are shown and compared with the base case.

The final section is a summary of orbital debris issues and contains the author's conclusions on how to best resolve the debris/policy problem.

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II. ORBITAL DEBRIS BACKGROUND

A. NATURAL DEBRIS

Since the beginning of time, there has always been orbital debris. In the early days it was in the form of meteoroids or natural debris. By definition, meteoroids are naturally occurring particulates associated with the solar system formation or evolution processes. This includes asteroid breakups and material released from comets. This type of debris is typically very small in size and has posed a remote threat to spacecraft because they have been strengthened to withstand the smaller sized meteoroids. The meteoroid is also a temporary problem because the average meteoroid will not assume an Earth orbit as it passes within gravitational range of the Earth. It will instead just make a pass by the Earth. With the meteoroids not remaining in an Earth orbit, the chance of a strike from natural debris is remote [Ref. 2].

B. ORBITAL DEBRIS

The other category of orbital debris is man-made and comes in many forms as will be discussed below. This form of debris possesses a higher threat because it will originate and stay within the Earth's orbit for a long period of time unless acted upon from external forces. Since this form of debris is our biggest challenge, it will be the focus of attention for this thesis.

More than 3,600 various space related missions dating back to the start of the space age era have left thousands of large, and even more smaller sized debris objects in near-Earth orbit. Since 1957, over 23,000 officially cataloged orbital debris objects have

been logged with almost one-third still in orbit [Ref. 2]. The term "cataloged" refers to objects larger than 10 cm in diameter that are trackable. With the new space station designed to survive an impact from orbital debris up to 1 cm, an area of concern is the estimate that there are 2 to 10 times as many 1-10 cm sized untrackable debris fragments as there are trackable objects. On average, since the start of the space age, the number of trackable (cataloged) objects has grown at nearly a net linear rate of 200 entries per year [Ref. 4]. With the only natural removal mechanism being atmospheric-drag, this yearly rate could pose a significant problem in future years.

With the growth rate of orbital debris growing each year, it is important to understand the origins of debris. Orbital debris can be put into four different categories - rocket bodies, fragmentation debris, non-functional spacecraft, and mission-related debris. Table 2-1 shows a perspective of the cataloged objects with respect to the altitude regime. Low Earth Orbit (LEO) clearly has the largest quantity of orbital debris.

Table 2-1. Cataloged Objects by Altitude Regime. From Ref. [5].

Altitude	Spacecraft	Rocket Bodies	Debris Fragments	Total
LEO	1292	3743	712	5747
MEO	107	3	24	134
GEO	465	3	133	601
Transfer	75	147	276	498
Other	359	229	361	949
Total	2298	4125	1506	7929

1. Rocket Bodies

Because of the immediate location of LEO, all spacecraft will either operate in, or transition through LEO. With LEO being a transition orbit, it receives the left over and spent rocket bodies. As shown in Table 2-1, rocket bodies are a very large problem both physically and numerically. Rocket bodies are typically left in LEO for some period of time. The presence of rocket bodies in orbit is of particular importance because of their characteristically large dimensions and of the potentially explosive residual propellants and other energy sources they may contain. Of the total debris population, one-sixth are derelict rocket bodies discarded after use. The larger stages, which are generally used to deliver spacecraft and any additional stages into LEO, usually reenter the atmosphere rapidly. The majority of orbital debris is the result of rocket body explosions [Ref. 6].

2. Fragmentation Debris

Fragmentation Debris is another large contributor to the cataloged Earth-orbiting space object population. Fragmentation includes debris created as a result of spacecraft collisions, explosions and/or the deterioration of a spacecraft or rocket body. These breakups are typically very destructive events that generate numerous smaller objects with a wide range of velocities. Although most fragmentation debris incidents have been accidental, some have been intentional breakups or explosions. See Figure 2-1 for a complete breakdown of breakups that have occurred. Of the 1506 total cataloged fragment objects that are in orbit, 712 are in LEO (See Table 2-1).

Collisions and deterioration contrast in that debris products from deterioration will typically separate smaller amounts of debris at low relative velocities and remain relatively close to the original spacecraft. A collision, on the other hand, will involve high relative velocities and extensive spreading of large amounts of debris in all directions. Delta velocity is defined as the change in the velocity vector caused by thrust measured in units of meters per second. When a collision occurs, a delta velocity is imparted to every object or piece of debris. Figure 2-2 shows the three types of delta velocities that occur as a result of fragmentation; normal, tangential, and radial. Whenever a fragmentation occurs as a result of a collision, it is reasonable to expect the resultant debris will incorporate all three types of delta velocity effects. The result is debris scattering in all directions (See Figure 2-2).

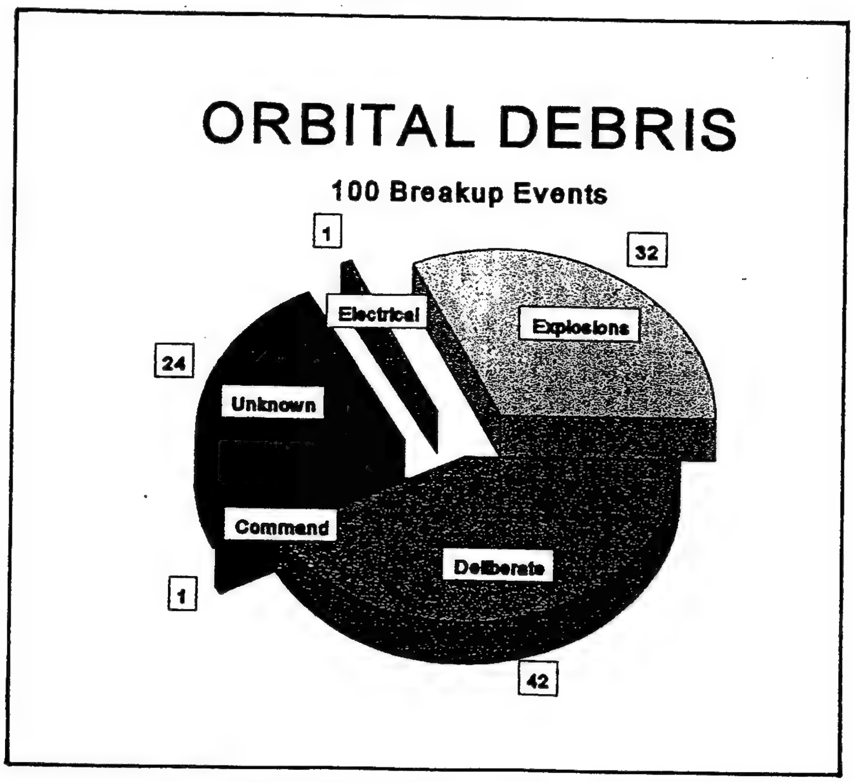

Figure 2-1. Breakup Events. After Ref. [1].

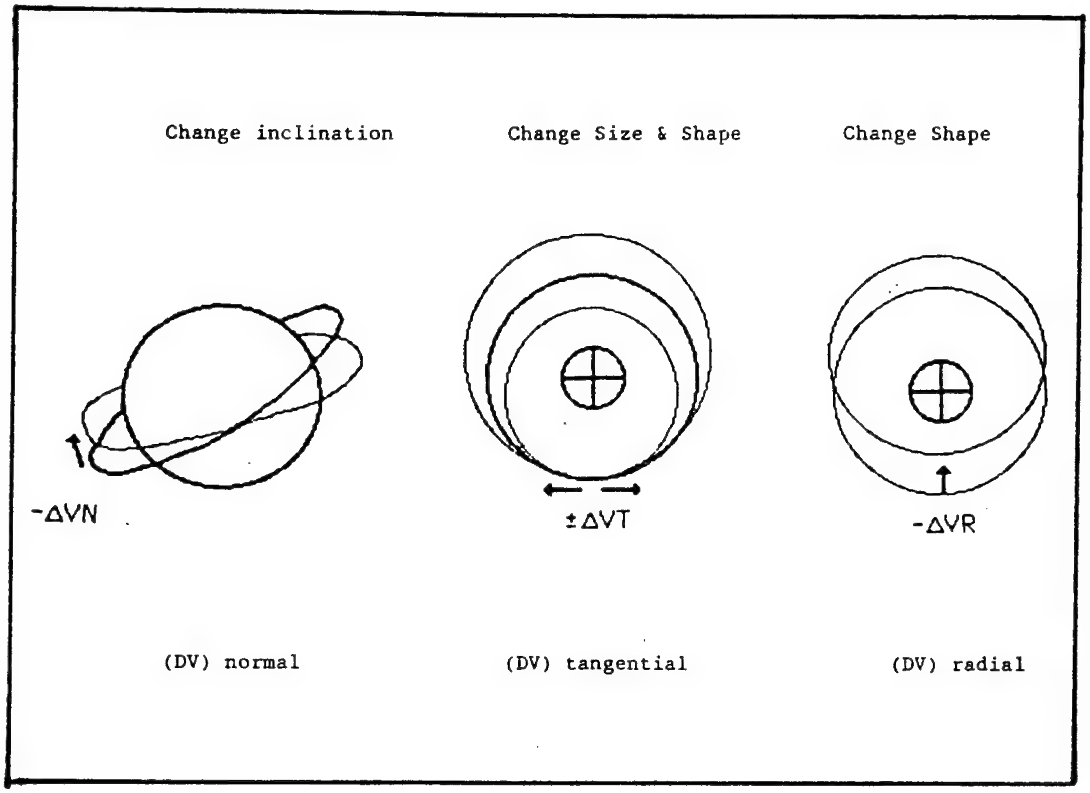

Figure 2-2. Effect of Delta Velocity. From Ref. [7].

The normal delta velocity causes a plane change and the radial delta velocty causes a change in eccentricity. These two delta velocities result in a relative motion which is periodic with a period equal to the orbit period. Thus, if the delta velocity only had these two components, the debris objects would remain close to the parent object. The tangential delta velocity has two effects. It causes a change in the period (semi-major axis) which results in the spreading of debris objects around the orbit as shown in Figure 2-3. The change of the semi-major axis causes a change in the right ascension precession rate. This causes a slow change in the orbital plane and creates a cloud (shown in Figure 2-3). Eventually a cloud will settle around the Earth with only the inclination controlling the width of the band.

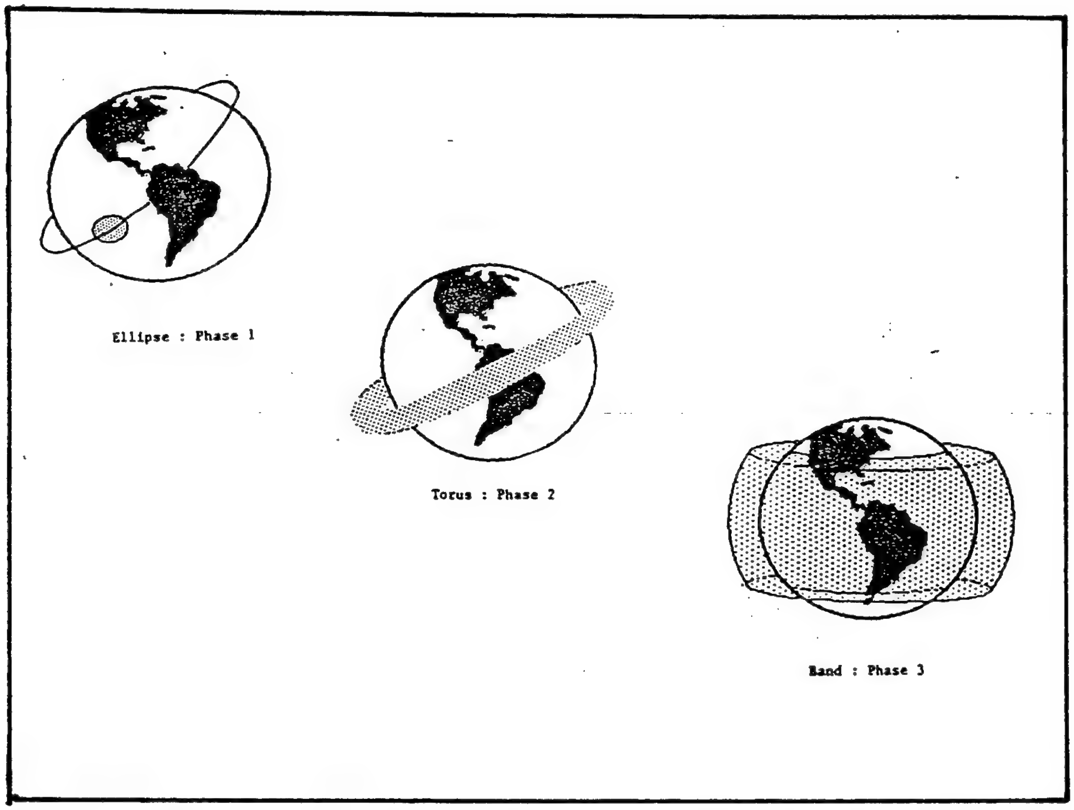

Figure 2-3. Phases of Debris Cloud Evolution. From Ref. [7].

Deterioration particles vary in size and come from thermal blankets, protective shields, solar panels, and include small paint chips. Despite the efforts in using the highest quality product, the severe temperature variations and radiation accelerates the aging of these products. Debris caused by deterioration will vary in sizes, with the majority between 0.1 and 1 cm (see Table 2-3). Of the total number of objects listed in Table 2-2, only the first row, or 8,000 objects are cataloged (trackable). Recently a piece of debris, a 3mm diameter piece of a circuit board, hit a shuttle bay door and remained embedded in the door. It caused no damage and is the first piece of returned debris.

Table 2-2. Estimated Debris Population. From Ref. [5].

Size	Number of Objects	% Number	% Mass
> 10 cm	8,000	0.02 %	99.93 %
1 - 10 cm	110,000 *	0.31 %	0.035 % *
0.1 - 1 cm	35,000,000 *	99.67 %	0.035 % *
Total	35,118,000 *	100.0 % *	2,000,000 kg #

* Statistically Estimated Values   # Calculated Value from reported data

3. Nonfunctional Spacecraft

Nonfunctional Spacecraft surprisingly represent four-fifths of the spacecraft population in Earth orbit. The remaining one-fifth are the operational spacecraft [Ref. 2]. When a spacecraft reaches its' end of life [EOL] point either through normal termination or a malfunction, it is usually left in the former orbit, or transferred to a slightly higher or lower orbit. The only exception to this accepted practice is the return of spacecraft in very low orbits to Earth upon termination of its mission. This exception has not been routinely done and was used more as a security measure than a debris mitigation measure.

4. Mission-Related Debris

Mission related debris includes anything that may be released as a result of a spacecraft's deployment, activation, or operation. Typical examples include pieces of explosive bolts, spring release mechanisms, or spin-up devices during a deployment phase. The MIR space station for example, had over 200 pieces of mission-related debris during its first eight years of operation. This is a classic example of the lack of attention or concern towards orbital debris. Along with the typical mission-related problems noted above, solid rocket motors present a problem equally as well. When a rocket is in a full

burn, large numbers of extremely small aluminum oxide (Al_2O_3) particles are formed and ejected through a wide range of flight path angles at velocities up to 4 km/s. Although these particles are extremely tiny, it is the actual number that is of concern. As many as 10^{20} may be created in a single rocket burn. To date there have not been any confirmed incidents in which collision with orbital debris has severely damaged or destroyed a spacecraft, but there have been a number of spacecraft malfunctions and breakups that might have been caused by impacts with debris. Examples of orbital debris issues involved the shuttle on two separate operational missions. During the first mission the cause of a chip on a shuttle's windshield was believed to be a paint flake. The second example was a recent shuttle mission that was on a routine mission and had to perform an emergency maneuver to avoid a non-functional spacecraft [Ref. 2].

C. DEBRIS DENSITY ISSUES

Currently the cataloged debris growth rate is linear averaging about 200 new items a year. At some point in the not too distant future, there is a probability that a major collision will occur and the debris growth will climb at an exponential rate from that point on. If this point is reached, it will become extremely difficult to operate in LEO if at all.

Within LEO, the largest population of debris is in the 700-1000 km band (See Figure 2-4) with an average concentration of about 100 objects in a 10 km altitude band. By comparison, at approximately 400 km where the Space Station International will orbit, the average concentration is about 10 objects in a 10 km altitude band [Ref. 4].

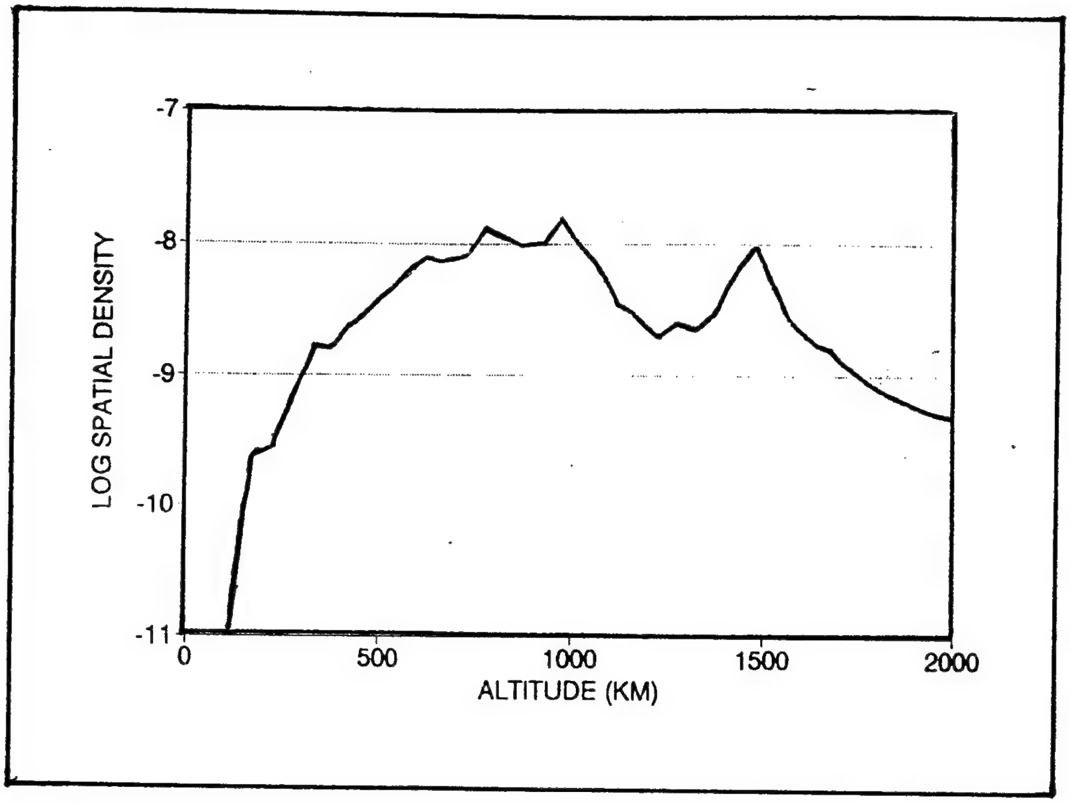

Figure 2-4. Spatial Density For Cataloged Orbital Debris. From Ref. [4].

Of the two major orbit regions, LEO and geostationary orbit [GEO], LEO is where the majority of the world's spacecraft operate. Several reasons drive why LEO has been the orbit of choice. First, it is far cheaper to launch to LEO using a much smaller launch vehicle than one that would be required for GEO. Second, the close proximity to Earth allows for remote sensing missions to receive higher resolution images. Third, the Earth's magnetic field protects spacecraft in some LEO's from cosmic radiation and solar flares. With humans operating in space, this last reason is of particular importance. The major LEO altitude band is from 700-1000 km. Based on NASA models, this altitude band is at or near critical density. Critical density, by definition, is when the debris population will produce fragments from random collisions at a rate that is increasing and is greater than the removal rate by natural processes [Ref. 8].

D. NATURAL METHODS OF DECAY

Currently the only natural method of removing orbital debris is atmospheric drag. As shown in equation [1] below, mass has a direct impact on the amount of drag imparted on a spacecraft. "The less massive the object for a given cross-sectional area, the greater its drag will be, resulting in a shorter lifetime in orbit" [Ref. 4]. This statement is shown to be true because of the location of mass in denominator.

$$A_D = -(½)\rho(C_D A/m)V^2 \qquad [1]$$

Where A_D = Acceleration due to Drag

C_D = Coefficient of Drag

A = Cross sectional Area

m = mass of spacecraft

V = velocity

ρ = Atmospheric density

Additionally, the 11-year solar cycle greatly affects the debris during peak cycles. The high solar activity heats Earth's atmosphere causing it to expand outward. This expansion increases the atmospheric density and, consequently the rate at which objects decay. The solar cycle effects are more effective with orbital debris below 600 km [Ref. 5]. Solar cycles do have a positive effect on the reduction of orbital debris as evidenced by two periods of decline between 1978-1981 and 1989-1991 which were solar maximum periods [Ref. 4]. As a perspective of satellite lifetimes in different orbits, Table 2-3 shows lifetimes with respect to orbit altitudes. Note the large difference in orbital lifetimes between 600 km and 1000 km.

Table 2-3. Examples of Lifetimes in Circular Orbit for an Average Satellite. From Ref. [4].

Altitude (km)	Lifetime (days or years)
200 km	1-4 days
600 km	25-30 years
1000 km	2000 years
2000 km	20,000 years

E. CHAPTER SUMMARY

Since the beginning of the space age, orbital debris in its many forms has grown at a linear rate. At some point in the future, this linear growth rate could go exponential with atmospheric drag currently the only real counteraction to debris growth. The growth of debris has been more concentrated in regions or bands of heavy use.

Attempts to estimate the turning point where the debris problem could go exponential is not easy. In fact accurate prediction of the growth rate is very limited due to the difficulty in observing small, very fast moving, and often dark objects against a dark background [Ref. 2]. Realizing the difficulties in accurate debris observation, all space faring nations should have a clear understanding that debris estimates and predictions could be in error with the actual situation better or worse than expected.

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III. HISTORY OF ORBITAL DEBRIS POLICY

Orbital debris has grown at a linear rate of about 200 new objects annually since the beginning of the space era. If the growth of debris is allowed to continue at this linear rate, there may be some point in the future where a major collision or explosion will occur causing the start of an exponential rate of new objects. Although when this might happen is unknown, what is crystal clear is the fact that space will become unusable under these conditions. The goal is to never hit that transition point where the orbital debris will go exponential. This problem has been identified for some time and efforts to educate and inform have been made. Some of these efforts have come by way of policy. These policies as a whole, have not been very restrictive or mandatory. Below are brief descriptions of the key policy documents regarding orbital debris.

A. OUTER SPACE TREATY

Not to long after the space age started, the **Outer Space Treaty** (10 October 1967) was the first written document governing the activities of States in the Exploration and Use of Outer Space, Including the Moon and Other Celestial Bodies. Under Article VIII of the treaty, launching states retain jurisdiction and control over their space objects. Objects included both operational spacecraft and orbital debris.

B. LIABILITY CONVENTION

Later in September of 1972, the **Liability Convention** established "fault" as a basis of liability for damage between space objects. It went on further to state absolute liability, allowing the injured party no need for proof of negligence or fault. The problem

is that both objects involved had to be unambiguous. Not only did the two objects in question have to be clearly identified, but the issue of "who hit whom" was equally important. This is very unrealistic as damage would most likely come from a tiny piece of debris that would be almost impossible to trace ownership. As the use of space increased, likewise the awareness of orbital debris issues increased.

In 1981, the AIAA published a Position Paper highlighting the very real hazards of orbital debris regarding on-orbit spacecraft, the potential permanence, and that there is no obvious or simplistic resolution. One of the most significant debris reduction efforts was the policy written by NASA in 1982 requiring the venting of all unspent propellants gases from Delta upper stages to prevent explosions due to fuel mixing. The Potential Threat to U.S. Satellites Posed by Space Debris was published in 1983 identifying no problem at that time, but continued research was recommended [Ref. 3]. Several years later, a serious attempt was made by several non-NASA American scientists to work together to consolidate the position on orbital debris. With the need for widespread awareness, several issues were identified. Those issues included [Ref. 3]:

a). An improved understanding of the debris environment is needed.

b). Understanding the physics of hypervelocity collisions is a must.

c). In order to better understand the debris environment the causes of satellite breakups must be known.

d). Research must be conducted to ensure the prevention of additional orbital debris and future spacecraft mass should be designed to withstand possible damage from orbital debris strikes.

C. INTERAGENCY GROUP - (SPACE DEBRIS)

In 1986, the **Interagency Group - Space Debris** was established specifically to address issues concerning orbital debris. Responding to the National Space Policy, the Interagency Group released a report that included the statement "...all space sectors will seek to minimize the creation of space debris. Design and operation of space tests, experiments, and systems will strive to minimize or reduce accumulation of space debris consistent with mission requirements and cost effectiveness." [Ref. 3]. Note that the policy is not mandatory and allows debris generation if mitigating debris is too expensive. Within the Interagency Group's report, several key findings and recommendations were made. Key findings were a) the growth of debris could threaten operations in space if left unchecked, b) little is known about small debris which resulted in an uncertainty in urgency for any corrective measures and c) the need for enhancing debris measurement. From those findings, several recommendations were made. They included 1) making debris minimization a design consideration, 2) emphasize and accelerate debris measurement, modeling, analysis of physical evidence from space, improved shielding technology, regulation development, and cost minimization, 3) DoD and NASA to work jointly in developing a plan for debris monitoring, modeling, and managing.

D. NATIONAL SPACE POLICY

Finally on 4 February 1987 the first real orbital debris policy was written. "DoD will seek to minimize the impact of space debris on its military operations. Design and operations of DoD space test, experiments and systems will strive to minimize or reduce accumulation of space debris consistent with mission requirements."[Ref. 3] The

following year on 5 January 1988, the **National Space Policy** was signed. It stated: "All space sectors will seek to minimize the creation of space debris. Design and operations of space tests, experiments and systems will strive to minimize or reduce accumulation of space debris consistent with mission requirements and cost effectiveness." [Ref. 3] On 16 November 1989, a sentence was added to the 1988 National Space Policy. It reads: "The United States Government will encourage other space faring nations to adopt policies and practices aimed at debris minimization"[Ref. 3].

E. USSPACECOM REGULATION 57-2

Two years later, the U.S. Space Command published the USSPACECOM REGULATION 57-2, a regulation addressing the minimization and mitigation of space debris. It specifically addresses the following responsibilities [Ref. 3]:

a. Through its component commands, USSPACECOM will foster activities to better understand the evolution of space debris and the hazards of orbital debris to military, civilian and commercial space activities.

b. Component space commands shall increase awareness of the requirement to mitigate space debris. They shall monitor space debris mitigation efforts of their material development activities, and, within their authority, assure that mitigation of space debris is addressed explicitly in all space systems developments and upgrades.

c. The design and documentation process for space system development, modification, or upgrade will permit clear identification of cost, schedule, and performance impacts of efforts to mitigate debris. System development or modification tradeoffs which affect the above in order to minimize debris shall be reviewed by and approved by the affected Service component space commands and coordinated with the United States Space Command.

d. The justification for measures to mitigate and minimize debris or the effects of hypervelocity impact upon space systems should reflect robust technical investigation and research. Component Commands shall focus research to quantify cost, schedule, and performance impacts on system development.

Within the different branches of the government, several working groups have been set up. NASA started the international working group, DoD started the Space debris working group, and DOT works closely with its contractors.

F. NASA MANAGEMENT INSTRUCTION (NMI) 1700.8

NASA published NASA Management Instruction (NMI) 1700.8 on 5 April 1993. This instruction applies to NASA Headquarters and Field Installations for all NASA programs/projects that may generate orbital debris and that become operational after the effective date of the instruction. NASA's policy is to employ design and operations practices that limit the generation of orbital debris with mission requirements and cost-effectiveness. For all NASA programs, orbital debris issues have and will be considered from the initial design phases.

NASA then followed up NMI 1700.8 with a handbook specifically written as guidance to limit the growth of orbital debris. The goal was to limit debris growth while at the same time minimizing extra costs. Eventually, after debris risk increases, more guidelines will be imposed with an increased cost. Several specific guidelines were outlined for program and project managers. They include [Ref. 9]:

a. Depleting on-board energy sources after completion of mission.
b. Limiting orbit lifetime after mission completion to 25 years or maneuvering to a disposal orbit
c. Limiting the generation of debris associated with normal space operations.
d. Limiting the consequences of impact with existing orbital debris or meteoroids.
e. Limiting the risk from space system components surviving reentry as a result of post mission disposal.

The guidelines set above were to prevent the orbital debris growth over the next 100 years while still minimizing the cost impact to spacefarers. Typically, upper stages and satellites with perigee altitudes below 600 km will successfully decay within 25 years. Satellites operating above 600 km will experience the largest impact with regard to this instruction because of the much larger natural decay lifetimes [Ref. 9].

G. SPECIAL COMMITTEE FORMED BY THE NATIONAL RESEARCH COUNCIL OF THE NATIONAL ACADEMY OF SCIENCES AND THE NATIONAL ACADEMY OF ENGINEERING

While NASA was publishing their handbook on orbital debris, a Special Committee was formed by the National Research Council of the National Academy of Sciences and the National Academy of Engineering to address the problems associated with orbital debris. This committee was tasked to characterize the current debris environment, project how it will change, explore ways to resolve the problem and develop a set of recommendations. Several options to slow the orbital debris growth rate have been considered and are shown in Table 3-1. The list of possible debris removal options continues to grow each year. Some examples of current concept plans include laser, sweeping, tether, solar sail, drag augmentation, retrieval, and propulsive maneuvers (deorbit). Using laser technology, a concentrated laser beam could be aimed at a piece of debris and completely vaporized or reduced to smaller less harmful pieces of debris. The concept of sweeping debris is a difficult one in terms of manufacturing and technology. The sweeping spacecraft will have to be able to withstand the impact of the orbital debris without creating more from those impacts. Of real concern with a sweeping concept is the fact that a sweeping must be able to identify orbital debris from operational spacecraft.

Another concept, the solar sail, is simple in principle. Solar radiation pressure would be used to change the orbital elements. This is a very slow process but would work effectively across all altitudes.

Table 3-1. Preliminary Debris Control Options from AIAA Survey and Community Discussions. From Ref. [9].

Mission Phase	Procedures Used
Launch/Deployment	Rocket Body: Reduce operational debris by use of bolt catchers (Delta and Centaur) Payload: Reduce operational debris by constraining lens covers, etc.
Operations	Mission Design: Avoid collision through the use of software tools (Shuttle) Payload: Mitigate effect of collision with addition of shielding (NASA)
Termination	Rocket Body: Vent excess propellants to prevent explosions (ESA, Atlas, and Delta) GEO Payload: Reorbit to a supersynchronous orbit (ESA, NOAA, and COMSAT) LEO Payload: Retrieval by the Shuttle Deorbit to lower elliptical orbit to accelerate decay time

Another similar concept to the solar sail is drag augmentation which takes the existing spacecraft or object and increases the physical area increasing the drag and decay time. The area of concern is when the cross-section or physical area is increased, it becomes a much larger target or object in space which will increase the probability of collision with another object. The increased probability of collision also applies to the solar sail design as well as the drag augmented concept as both would have a large cross sectional areas.

The retrieval concept is to fly another spacecraft to the object or debris, attach, and produce a orbit change decreasing the total decay time. To date retrieval has only been done with manned spacecraft (Shuttle).

The propulsive maneuver technique includes using thrusters (possible existing station keeping thrusters) to move the satellite into a lower orbit, reducing the total time of decay. This technique will be the method used in the satellite model in the following chapter.

H. CHAPTER SUMMARY

Debris control options can be categorized into two groups, prevention and removal. The author will focus on the removal or deorbiting of LEO satellites only. When considering the effect of the Earth's gravitational field, it is more economical to deorbit a satellite towards the Earth when the satellite altitude is initially below 25,000 km.

The awareness and understanding for the orbital debris problem continues to grow. The overall agreement is to work on the prevention of more debris while trying to develop cost effective ways to conduct orbital debris removal [Ref. 4]. Since current technology is not cost effective for removal of existing debris, the focus of this paper will be on the removal of satellites at the completion of mission life.

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IV. ORBITAL DEBRIS MITIGATION ANALYSIS

Now that a brief history of orbital debris and the current applicable policies have been reviewed, the next step is to address the cost impact of alternative spacecraft design options that support debris mitigation. The cost impact will be shown by first developing characteristics of a base case (generic) satellite and applying it to several different debris mitigation options.

A. DESCRIPTION OF THE STUDY

1. Problem

Orbital debris will continue to be a problem as long as cost is a factor in the decision process. Minimizing lifecycle costs while mitigating debris is a universal problem that will not be easily solved. This thesis will address the cost impact issues by providing some design alternatives to help resolve this cost impact problem.

2. Solution Approach

The design options that were developed represent current, short-term future, and long-term future. Using the base case (generic) satellite, these design alternatives are developed, showing the cost impact with respect to debris mitigation.

a. Option I

Under Option I, a satellite would fully complete its operational mission and then be left on-orbit to decay naturally. This natural decay is not a debris mitigating practice. It is, however, the current method of operating. This option most closely reflects today's practices.

b. Option II

Under Option II, a satellite would be deorbited prior to its full mission completion period in order to complete a successful deorbit. A successful deorbit is a decay lifetime of 25 years or less. This satellite has the same characteristics as the model used in Option I. The concept of Option II is a retro-fit or quick-fix to an existing system.

c. Option III

Under Option III, a redesigned set of satellite characteristics is used. This redesign allows enough fuel for a complete operational mission period and a successful deorbit. For the purposes of this thesis, the redesign only addresses a larger fuel load. In each design alternative that will follow, Option III will always have the required fuel for deorbit available. The concept for Option III is that it is future planning and designing.

B. GENERIC SATELLITE SYSTEM CHARACTERISTICS

Throughout all altitudes of LEO, 25% of the satellites currently on-orbit have a communications mission. Additionally, 1082 more communications satellites are projected for launch to LEO altitudes over the next several years (see Table 4-1). LEO is the most popular operating region. This is due in a large part to launch costs and mission requirements.

Because of this increased interest in communications satellites operating in LEO, the generic satellite was developed to resemble the most reasonable type, size, and mission expected to be launched into LEO in the near future, a communications satellite. Table 4-2 gives a break down of the specific parameters and constraints established for the generic satellite model. In Table 4-2, the on-orbit altitude parameter shows two

Table 4-1. Some Proposed LEO Constellations. From Ref. [5].

System	Number of Spacecraft	Altitude (km)	Inclination
Teledesic	840	700	98.2
Iridium	66	780	86.0
Globalstar	48	1400	47.0
Odyssey	12	10360	55.0
Aries	48	1020	90.0
Ellipsat	24	500-1250	63.5
Vita	2	800	99.0
Orbcom	18	970	40.0
Starsys	24	1340	50-60

values, 800 km and 1300 km. This is because the model will be "flown" and compared at both altitudes to show the impact altitude has with mitigating orbital debris. The generic satellite will be evaluated in terms of decay lifetime and lifecyle cost for each option evaluated. This chapter will show a comparison of the different options with fixed parameters. A comparison of the different options will show the cost impact with respect to the decay lifetime.

This generic satellite model will be used for all three options that have been identified above. Note that option three will include the same satellite with a larger load of fuel.

Table 4-2. Satellite Model Parameters and Constraints. After Ref. [10].

Parameter/Constraint	**Value**
Mission type	Communications
On-Orbit Altitude (Circular)	800 km and 1300 km
Inclination	81°
Cross-Section	6 m^2
Deorbit Perigee Kick Altitude (Elliptical)	Various altitudes
Spacecraft Dry Weight - *Structure* - *Thermal* - *ADCS* - *Electrical Power System (EPS)* - *Tracking, Telemetry and Control (TT & C)* - *Communications* - *Apogee Kick Motor (AKM)* - Total Spacecraft Dry Weight	 300.8 lbs 30 lbs 75 lbs 631 lbs 30.6 lbs 192 lbs 440.6 lbs 1700 lbs or 771.1 kg
Total Fuel Weight	300 lbs or 136.1 kg
Satellite Design Life	5 years
Operational Mission Period	10 years

C. ANALYSIS OF OPTION 1

Under Option 1, a satellite would fully complete its operational mission and then be left on-orbit to decay naturally. The following analysis will show natural decay lifetime and the system lifecycle cost. System lifecycle cost is defined as the total costs of launching and operating one or more satellites (as required based on the alternatives) to cover the ten year operational mission period.

1. Decay Time (Natural)

Starting with the base case model at 800 km and 1300 km circular orbits, the first step is to compute the natural time to decay from those orbits. Natural decay by definition is the period of time a satellite takes to leave its initial orbit and "free-fall" without any assistance (from the satellite) to the Earth's atmosphere where the satellite is burned up on re-entry or falls to the surface. Natural decay is primarily caused by atmospheric drag. As will be shown later in this section, the difference in decay times between 800 km and 1300 km is large. This large difference in decay lifetimes is a result of the reduced impact atmospheric drag has with increasing altitude.

There are several methods for estimating satellite decay time. Due to the time varying uncertainty of the atmoshpere, the accuracy in the lifetime estimate is rarely better than ± 10 %. For a best estimate, one should use numical integration techniques that use the the best estimate of the time varying atmospheric density and take into account lunar and solar perturbations. Numerical integration is impractical for all satellites and approximate methods are needed. This thesis uses the method developed by King-Hele [Ref. 12]. This method uses the average value of the atmospheric density. For long lifetimes (much longer than the 11 year solar cycle so that the averaging of the density over the solar cycle is valid) this approximate method is typically within 2 % of the more accurate numerical integration techniques. Figure 4-1 is the lifetime decay graph that was developed by King-Hele [Ref. 11] thats based on the mean density over an average solar cycle. This graph is based on mean density over an average solar cycle. Total decay time in days is shown as (L) with (S) representing the average cross-sectional

satellite in square meters, and (m) representing the mass of the satellite in kilograms. Enter Figure 4-1 with the perigee height and eccentricity. Since the 800 km and 1300 km orbits are circular, perigee will equal apogee resulting a zero eccentricity. Start at the bottom of the Figure, enter with the perigee altitude and travel vertically to the

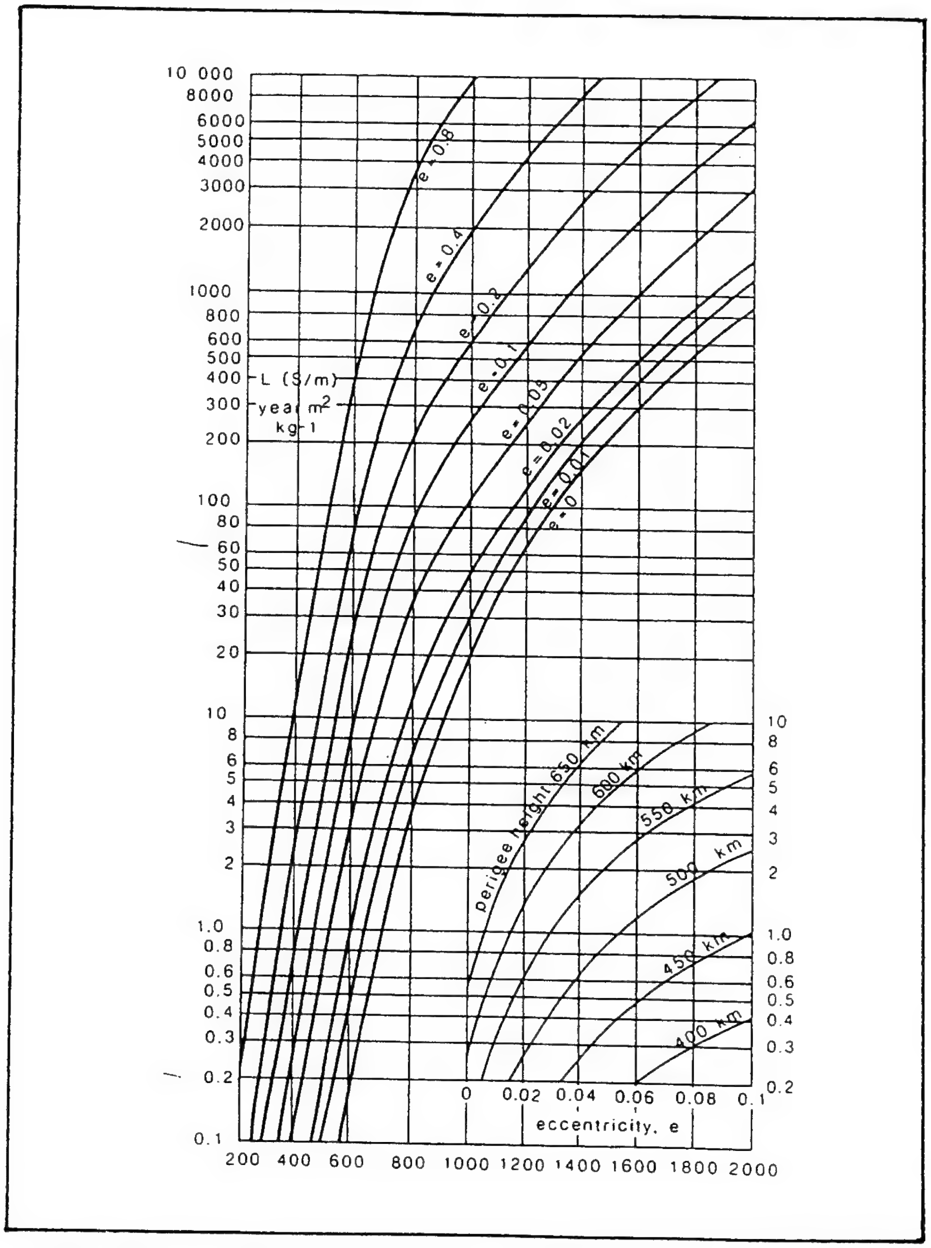

Figure 4-1. Lifetimes of Long-lived satellites, based on mean density over an average solar cycle. From Ref. [11].

"e=0" curved line (first one in this case). At the intersection of those two values, move horizontally across to the left side of the graph and read the L(S/m) value. Note that the left side of the chart is in log format. Take the extracted L(S/m) value and multiply with (m/S).

$$\text{Decay Time (years)} = L(S/m)*(m/S) \qquad [2]$$

where m = mass of satellite in kg

S = the average cross-sectional area of satellite in m^2

L(S/m) = Value extracted from left side of Figure 4-1

Using the values of S and m from Table 4-2, the satellite model would take 574 years to decay naturally from an initial altitude of 800 km. Starting from an initial altitude of 1300 km, the decay dramatically increases to 15,872 years. Additionally the satellite model was put into the SATRAK [Ref. 13] software program to determine decay times. The results from the SATRAK software program were within one to two percent of the King-Hele graphical method.

King-Hele's decay method can be worked backwards. For example, if the decay lifetime was known and the required perigee was the needed value. In this example, take a lifetime requirement of 30 years, enter Figure 4-2 below. At the 30 year decay time, a perigee of 485 km will be required for the 800 km apogee model and a 422 km perigee for the 1300 km apogee model.

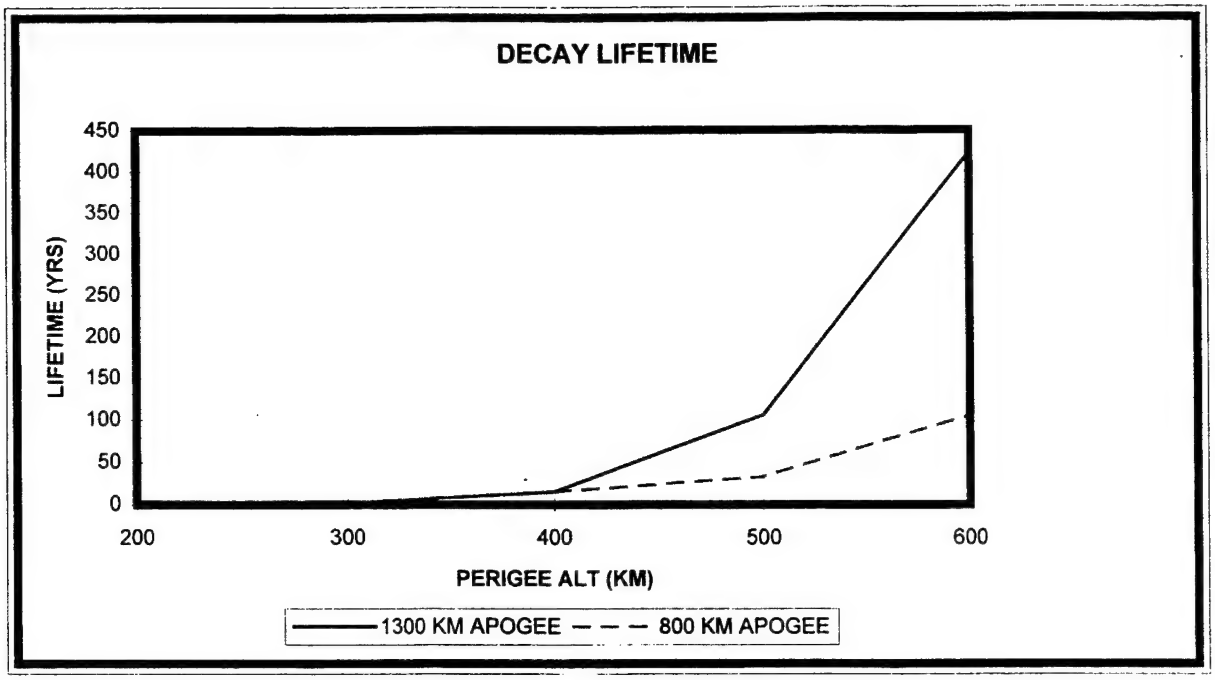

Figure 4-2. Decay Lifetimes for 800 km and 1300 km Apogee Models.

2. Probability of Collision

With the decision of a company to exercise option one (natural decay), the next question is what is the probability of that satellite colliding with another satellite? The probability of collision is mainly a function of the spacecraft size, the orbital altitude, and the period of time that the spacecraft will remain in orbit. To compute the probability of a collision (PC), the following is given [Ref. 3]

$$PC = 1 - \exp[-(SPD)(AC)(VREL)(T)] \qquad [3]$$

$$= (SPD)(AC)(VREL)(T), \text{ for small PC values}$$

Where PC = Probability of Collision (no units)

VREL = Relative Vel. between satellite and debris (km/sec)

(Average relative velocity is 10 km/sec)

AC = Cross-Sectional Area of satellite at risk (km^{2})

SPD = Spatial Density (number of objects per cubic km)

T = Time interval (sec)

After calculating the base case parameters through the Probability of Collision equation, the actual probability of collision with trackable object is much less than 0.01 per year. Said another way "...For a 20 square meter cross section satellite at 850 km, the probability of a collision with a trackable object is 1:10,000 per year. An operational satellite in this region will have a 99.9% probability of surviving a 10 year mission without being struck by a cataloged object."[Ref. 4] Also listed below in Table 4-3 are probability of collision values provided as perspective of the actual remote chance of a major collision with a trackable object.

Table 4-3. Probability of Collision values for Representative Satellites. From Ref. [5].

Satellite	Cross-Sectional Area	Average Altitude	PC/Year
MIR Space Station	270	350	2.0×10^{-4}
GEOSAT	32	790	1.1×10^{-4}
Landsat 4	37	700	7.6×10^{-5}
Solar Mesosphere Exp	2.6	500	4.5×10^{-6}

3. Lifecycle Cost: Option One

Typically, costs can be estimated using the engineering buildup, analogy, or parametric method. The engineering method starts at the absolute lowest level of design and works its way upward covering every single detail. As will be explained in more detail later, a generic satellite model is going to be used to help show the cost

comparisons. Because this model does not represent a specific system, the engineering buildup is far too detailed for the purpose of the model. Next, the analogy method uses direct comparisons between similar systems. Again, since generic satellite characteristics are being used, no direct comparisons could be effectively made. This makes the analogy method not a good choice. The last method, parametric, is more of a broad scope and generalized look at cost estimates. Because parametric is more generalized, it will be the method of choice. This parametric method takes statistical formulas and historical cost data and merges them to form cost estimating relationships (CER's).

The most important factor in using the parametric method is to establish the correct CER's that represent the cost model being estimated. The generic satellite characteristics represent a small satellite with an approximate dry weight of 1700 lbs. Because of the small satellite size, the Small Satellite Cost-Estimating-Relationship was initially used [Ref. 10]. However, because the majority of the satellites in the database were not similar in mission type and the weight was much less than the generic satellite used in this thesis, it did not provide the best CER's and therefore was not used. Instead, the Unmanned Space Vehicle Cost Model (Edition 7) was used [Ref. 10].

Within the Unmanned Space Vehicle Cost Model handbook, there are two types of CER's that can be used. For the generic satellite, the minimum percentage error (MPE) CER's will be used. These CER's, once summed up, will provide total recurring and total nonrecurring costs. A summary of these values is shown in both Tables 4-4 and 4-5 with the specific computations listed in greater detail in Appendices B and C.

Table 4-4. Recurring Cost Estimates Using USCM7. After Ref. [10].

Spacecraft Subsystem Elements	Recurring Cost ($ in thousands)
Structure	1756.1
Thermal	312.1
ADCS	
- Attitude Determination	4111.4
- RCS	1575.3
Electrical Power Supply	
- Generation	874.7
- Storage	1111.8
- PCD	3139.2
Telemetry, Tracking and Command	
- Transmitter	119.8
- Transmitter	140.3
- Receiver/Exciter	509.8
- Transponder	630.4
- Digital Electronics	2353.1
- Analog Electronics	286.6
- Analog Electronics	489.1
- Antenna (Horn)	842.7
- Antenna (Dipoles)	21.8
- Antenna (S-Band)	24.9
- RF Distribution	92.5
Communications	
- TWTA	305.3
- Solid State	1965.9
- Receiver/Exciter	1097.5
- Transponder	2023.0
- Transponder	1449.8
- Digital Electronics	1291.2
- Antenna	1092.7
- Antenna Reflectors	530.1
- RF Distribution	283.4
Apogee Kick Motor	346.8
IA&T	8216.1
Total Space Vehicle	36993
Program Level	1069.1
LOOS (3-Axis)	3760.4
Total	41822.5 (41.8 M)

Table 4-5. Nonrecurring Cost Estimates Using USCM7. After Ref. [10].

Spacecraft Subsystem Elements	Nonrecurring Cost ($ in thousands)
Structure	8918.28
Thermal	2773.22
ADCS	
- Attitude Determination	9981.47
- Reaction Control System	1524.38
Electrical Power System	
- Generation	7002
- Storage	1276.93
- PCD	6618
Telemetry, Tracking and Command	
- Transmitter	349.03
- Receiver/Exciter	545.76
- Digital Electronics	4142.19
- Antenna	2314.50
Communications	
- TWTA	4272.51
- Solid State	4845.53
- Receiver/Exciter	3586.69
- Transponder	3297.47
- Digital Electronics	10152.08
- Antenna	2866.56
- Antenna Reflector	14111.11
Total Space Vehicle	88577.71
IA&T	17874.73
Program Level	23262.37
Age	11903.66
Total	141618.47 (141.6 M)

After computing all the CER's for each subsystem and totaling the amount, the total recurring cost estimate for the base case is $ 41.8 M (FY92$) (see Table 4-4). The total nonrecurring cost estimate for the base case is $ 141.6 M (FY92$) (see Table 4-5). These two cost estimates combined with a 10 % discount rate will serve as the foundation numbers in ultimately computing the Present Value of Satellite Lifecycle Cost.

The Present Value of System Lifecycle Cost used in this thesis represents the sum of satellite lifecycle and launch costs. Satellite lifecycle cost is the sum of present value of recurring costs and present value of noncurring costs. Nonrecurring costs are associated with all the effort and activity of designing, developing, manufacturing, and testing a space qualified model or system. To convert nonrecurring costs to Present Value of Satellite Nonrecurring Costs, discounting must be applied. Equation [4] below will be used in the following analysis for computing the Present Value of Satellite Nonrecurring Costs.

$$PVSNR = NR/(1 + dr)^t \qquad [4]$$

Where PVSNR = Present Value of Satellite Nonrecurring Costs

dr = Discount rate (10 %)

t = The year the satellite was manufactured

NR = Nonrecurring Cost ($ 141.6 M from Table 4-6)

Recurring costs on the other hand are associated with all of the efforts connected with continuing orbital and terrestrial operations. Converting the recurring costs to a Present Value of Satellite Recurring Costs also requires the application of discounting. Equation [5] is used in computing the Present Value of Satellite Recurring Costs.

$$PVSR = R/(1+dr)^L+...+R/(1 + dr)^t...+R/(1+dr)^T \qquad [5]$$

Where PVSR = Present Value of Satellite Recurring Costs

R = Recurring Costs ($ 41.8 M from Table 3-5)

t = The year the costs were incurred starting with t=L (year of launch) and continuing to t=T (end of the planning period)

The next cost that has to be added to the total satellite cost is the launch cost. The launch cost will vary depending on the launch vehicle used. For this example, the Taurus Launch vehicle has been chosen because of its ability to launch the satellite model to both altitudes used in this simulation (see Figure 4-3 for Taurus performance chart). The estimated cost for a launch from a Taurus Launch Vehicle is $15,000/kg [Ref. 14].

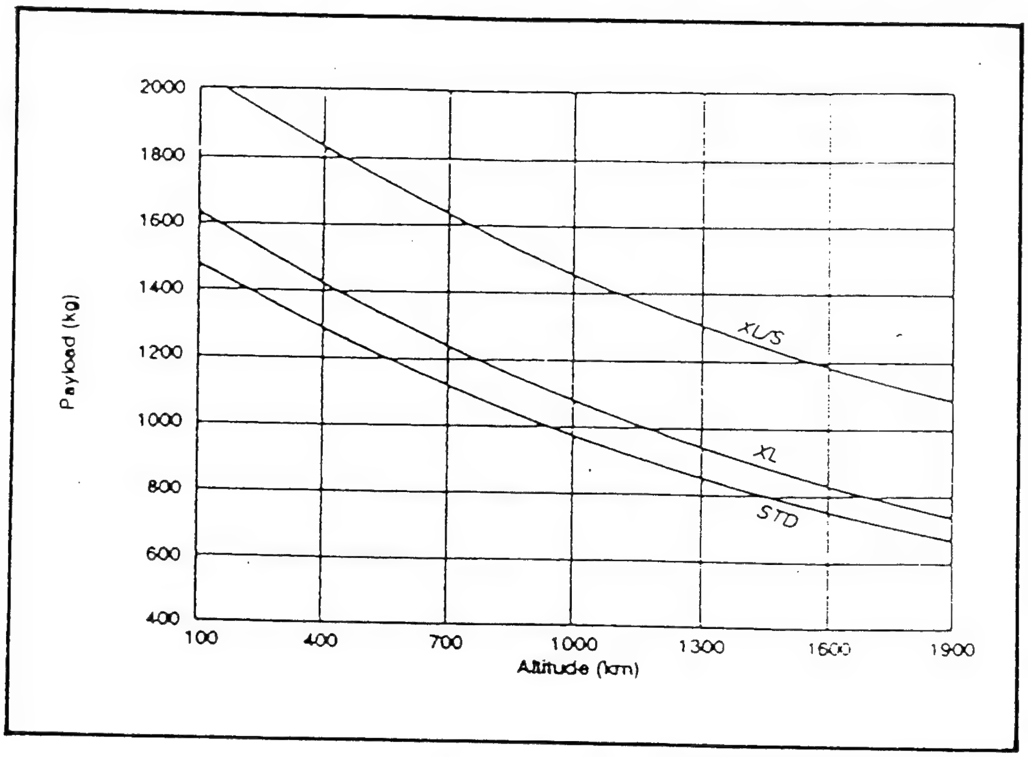

Figure 4-3. Taurus Performance Chart. From Ref. [14].

With a 2,000 pound satellite, the launch costs equal $13.6 M for a launch to either 800 km or 1300 km. Referring back to Table 4-2, a 10 year operational mission requirement was established and the generic satellite has a mission design life of five years. In order

to fully complete the 10 year operational period, two satellites will be required. The two satellites will cover the 10 year period exactly with each satellite expended completely after each five year period. Throughout the different options and analysis alternatives that will follow, it is assumed that the generic satellite will have a 100 % performance rating, requiring no need for mission spares. Therefore mission spares will not be addressed in this thesis. The cost of the second satellite launched at the five year mark was discounted and adjusted back the to present. With equations [4] and [5] above and the requirement for two satellites, Lifecycle Cost can now be computed. Since there are two satellites, the PVSR will have to be computed for each one and then added together. The first satellite PVSR will be labled PVSR(1) and the second satellite PVSR(2). This label method will continue throughout this thesis. Using equation [3], PVSR(1) equals $ 173.6 M (see Table 4-6). The second satellite, PVSR(2) will start at t=5 (five year point) and progress to the t=10. The next step is to compute the PVSNR. This, unlike the previous PVSR, can be accomplished in one equation. Using equation [5] from above, insert the values for this case (Table 4-7 shows the work).

Table 4-6. Calculations for Present Value of Recurring Costs Using Option I.

PVSR(1) = 41.8+(41.8/1.1)+(41.8/1.21)+(41.8/1.3)+(41.8/1.5) = 173.6
PVSR(2) = (41.8/1.6)+(41.8/1.77)+(41.8/1.95)+(41.8/2.14)+(41.8/2.36) = 108.3
Total PVSR = $ 281.9 M

Table 4-7. Calculations for Present Value of Nonrecurring Costs Using Option I.

PVSNR = 141.6 + (141.6/1.6) = $ 230.1 M Note that the series contains the same number of terms as satellites. Also note that the second satlellite has been discounted for the year of launch (t=5).

Combining the total PVSR and PVSNR, the Satellite Lifecycle Cost (LCC) can be obtained. The Satellite LCC in this case is $ 512 M. The last step is to add the launch cost to the Satellite LCC. With the requirement of two launches and the cost at $ 13.6 M each and discounted for the year of launch, $ 22.1 M will need to be added to the Satellite LCC. This results in a System LCC for Option of $534.1 M. Because of the repeated number of times LCC will be computed for the remaining cases, the steps be much more brief with the work shown for each case.

OPTION ONE LIFECYCLE COST	$ 534.1 M

D. ANALYSIS OF OPTION 2: DEORBIT AN EXISITING DESIGN PRIOR TO MAX LIFETIME ON ORBIT

A more aggressive approach to mitigating orbital debris is to deorbit a spacecraft instead of leaving it to decay naturally. This is especially important for satellites operating full-time in high traffic altitudes such as the 700-1000 km band. As would be expected, there are some trade-offs in deorbiting a spacecraft. Because of the forecasted increase in the use of LEO altitudes, some companies are taking the aggressive approach

to minimize orbital debris [Ref. 6]. The base case for Option 2 takes the same generic satellite used in Option 1 and applies an aggressive approach to helping reduce orbital debris.

1. Decay Time: Deorbit Prior to Completion of Design Life

The method of debris mitigation used in this example is to deorbit the satellite, decreasing the total decay time. The deorbit will consist of taking the satellite from a circular orbit to an elliptical orbit with the perigee altitude low enough to help increase the rate of decay. The technique is to lower perigee and let drag lower apogee. This maneuver could remove the spacecraft from a higher risk orbit immediately and generally put it into a lower risk orbit until full decay occurs.

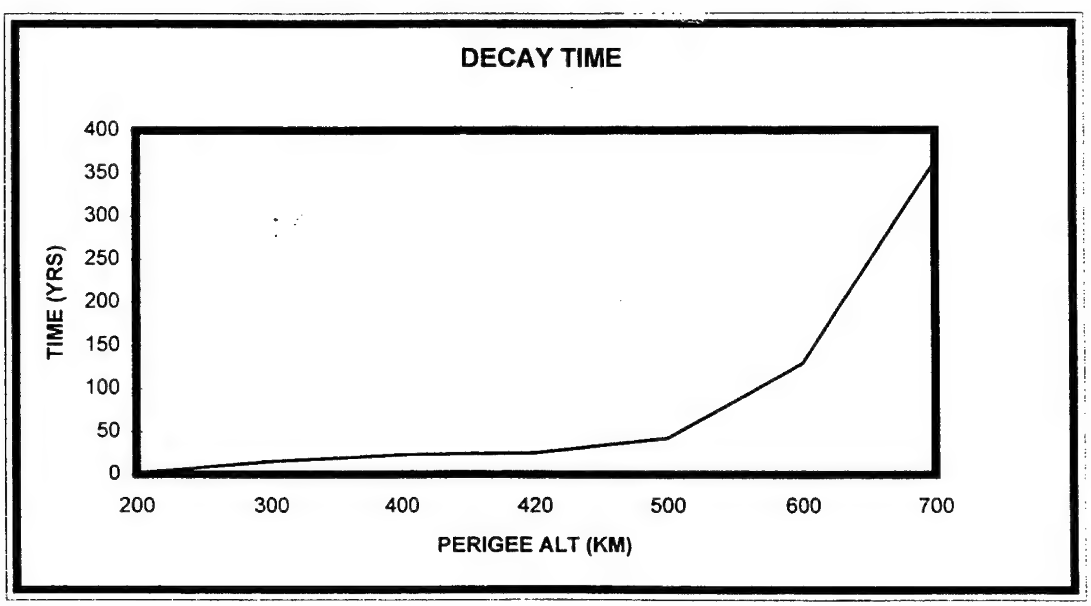

Figure 4-4. Decay Time. 800 km Apogee Altitude.

In determining the decay time, Figure 4-1 and equation [2] will be used again. The decay lifetime curve for both the 800 and 1300 km apogee altitudes is graphically shown in Figures 4-4 and 4-5. A specific altitude and decay time will be selected in the next section when comparing the fuel requirements with altitude.

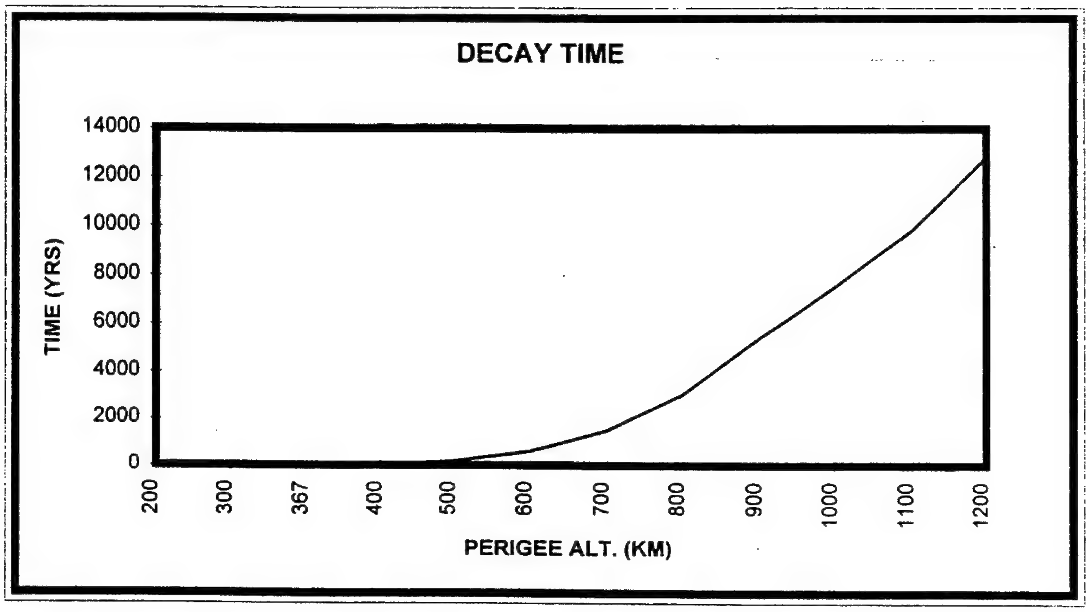

Figure 4-5. Decay Time. 1300 km Apogee Altitude.

2. Fuel Requirements For Satellite Deorbit

In order to determine the actual dollar cost it will take to deorbit the spacecraft, ΔV has to be computed first. Start with computing the velocity (V) of the satellite in its original orbit

$$V^2 = \mu/R_1 \qquad [6]$$

Given that: $\mu = 3.986 \times 10^5\ km^3/s^2$

$R_\oplus = 6378.14$ km

r_1 = Spacecraft altitude

$R_1 = R\oplus + r_1$ [Apogee Altitude]

Now compute the ΔV (See Appendix A for specific computations):

$$\Delta V = [V]^{1/2} [1-(1-e)^{1/2}] \qquad [7]$$

Where e = the eccentricity of the new orbit (see Appendix A)

Taking the ΔV from above, apply that value to the following equation to solve for the change in mass (ΔM)

$$\frac{\Delta M}{M_o} = 1-\exp[-\Delta V/g\, I_{sp}] \qquad [8]$$

Where M_o = Initial Mass (2000 lbs or 907.2 kg in this example)

ΔV = km/s (from above)

I_{sp} = 250 sec and 300 sec

Therefore ΔM equals the total fuel required to transfer the base case to the perigee altitude selected. Figure 4-6 shows fuel required with respect to perigee. See Figures 4-6 and 4-7 for the amount of required fuel as a function of perigee.

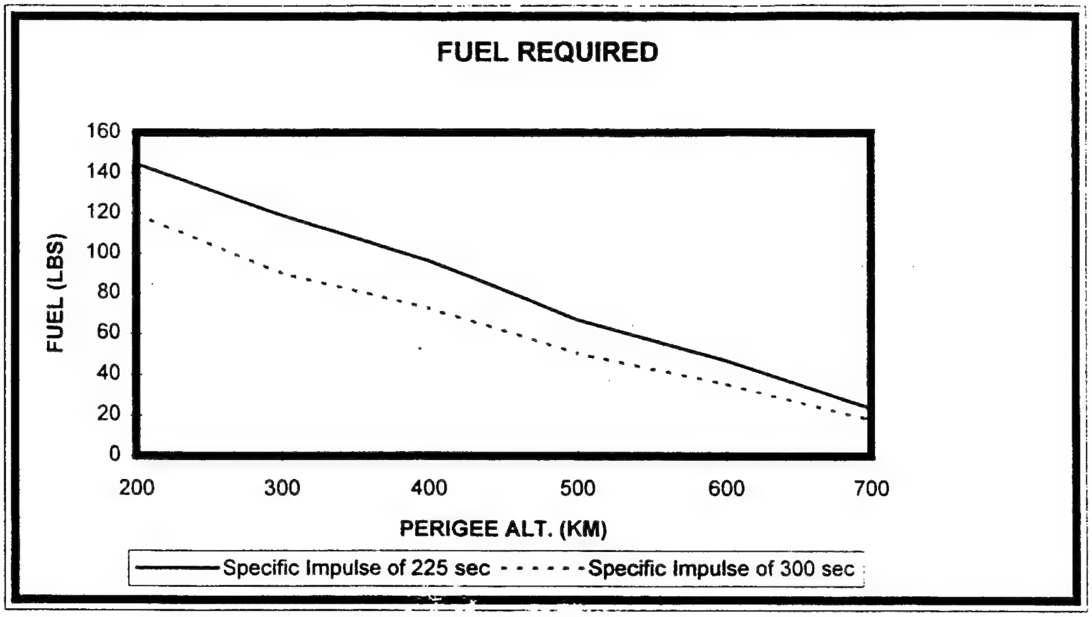

Figure 4-6. Fuel Required, 800 km Apogee Altitude.

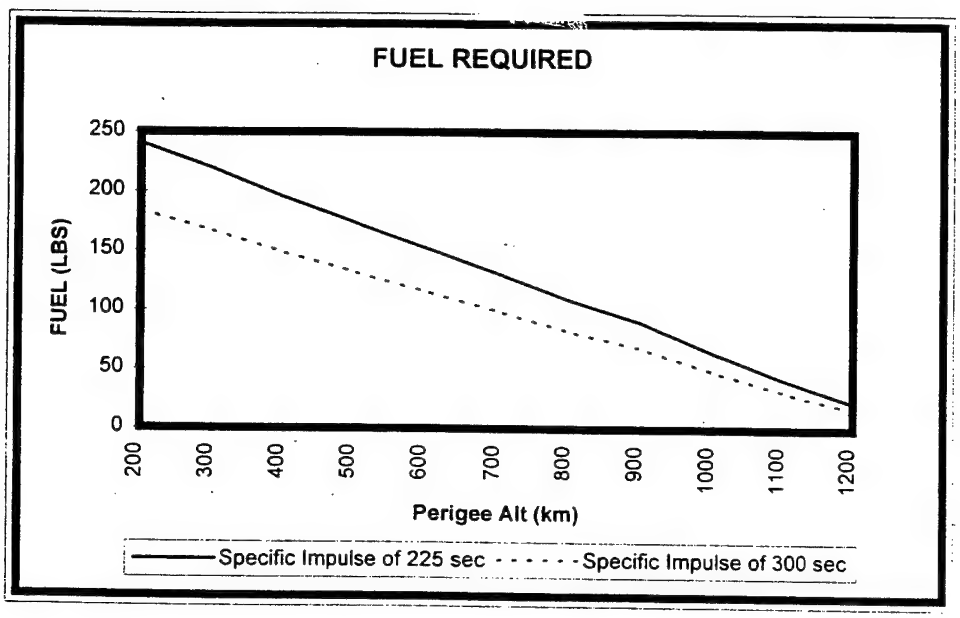

Figure 4-7. Fuel Required, 1300 km Apogee Altitude.

The total fuel load for the satellite model is 300 pounds. This amount has been established based on satellite size and mission. Two thirds (200 lbs) of that will be used for establishing the satellite on-orbit and the remaining one third (100 lbs) will be used for on-orbit station keeping. With 100 lbs of fuel to cover station keeping for the entire design life of 60 months (five years), the average amount of fuel required per month is 1.667 lbs of fuel. Examining Figures 4-4 and 4-5, the decay time varies significantly with altitude showing the most dramatic changes between 600 km and 1000 km. NASA, through the Interagency Group, just released Reference 2 which establishes NASA's guidelines regarding total decay time. NASA has set a limit of 25 years for the full decay of all satellites. In keeping with the goals of option two, mitigating debris, NASA's 25 year limit will establish the required perigee altitudes from Figures 4-4 and 4-5. Using Figure 4-4 and an entering with the 25 year limit, a perigee of 465 km is required for an 800 km apogee. For the 1300 km apogee, the required perigee is 367 km (see Figure 4-5).

Once the perigee altitude has been established, the amount of required fuel can now be computed. Figures 4-6 and 4-7 show the fuel required with respect to perigee altitude for both apogee altitudes. Included in both figures is a comparison between a specific impulse (I_{sp}) of 225 seconds and a specific impulse (I_{sp}) of 300 seconds to show increased fuel requirement changes with respect to specific impulse types. The assumption for the base case is a specific impulse of 300 seconds. Entering Figure 4-6 with a perigee of 465 km and an apogee of 800 km, 60 lbs of fuel will be required to deorbit the satellite in 25 years. Next, entering Figure 4-7 with a perigee of 367 km and

an apogee of 1300 km, 155 lbs of fuel will be required to deorbit the satellite in 25 years. With only 100 lbs of fuel available, this is not possible. Using Figure 4-7 again, enter the left side of the graph with 60 lbs of fuel and read off the perigee altitude at the bottom. The result is approximately 900 km. Next go to Figure 4-5 and enter that graph with the new perigee of 900 km. The new decay time is approximately 5,442 years.

3. Lifecycle Cost: Option Two

The first step for Option II cost is to take the required deorbit fuel from above and convert into actual months of coverage. Take the required fuel for a successful deorbit (60 lbs) and divide it by the average amount fuel used per month (1.667 lb per month). This result of 36 months is the amount months lost per satellite as this fuel will be used instead for deorbit. Since the satellites have a design life of 60 months (5 years), 24 months will be available for on-orbit operations. In order to complete the ten year operational mission requirement, satellites will have to be replaced every 24 months, totalling five satellites. As a reminder, the generic satellites are assumed to perform perfectly, requiring no need for mission spares. Once the required number of satellites is established, the System LCC can be computed. Just as in Option I's cost analysis, the PVSR and PVSNR will be computed the same. Note that there are more satellites which will increase the series in the same equations. Table 4-8 below shows the work for Option II. Again, adding Satellite LCC and the present value of Total Launch Costs (TLC) together results in the System LCC. In this case, the System LCC is $ 829.05 M.

Table 4-8. Calculations for Lifecycle Costs Using Option II.

PVSR(1) = 41.8 + 38 = 79.8 PVSR(2) = 34.5 + 31.4 = 65.9 PVSR(3) = 27.9 + 26.1 = 54 PVSR(4) = 23.6 + 21.4 = 45 PVSR(5) = 19.5 + 17.7 = 37.2	Total PVSR = $ 281.9 M
PVSNR = 141.6+(141.6/1.21)+(141.6/1.5)+(141.6/1.77)+(141.6/2.14) = $ 499.2 M	
Satellite LCC => PVSR + PVSNR = $ 781.1 M	
Total Launch Costs (TLC) => (13.6) + (13.6/1.21) +(13.6/1.5)+(13.6/1.77)+(13.6/2.14) = $ 47.95 M	
System LCC for Option II => Satellite LCC + TLC = 781.1 + 47.95 = $ 829.05 M	
OPTION TWO SYSTEM LIFECYCLE COST	$ 829.05 M

Comparing Option II with Option I shows the impact of decay lifetime tradeoffs. Option II has a decay lifetime of 25 years at an increased cost of $ 294.95 M more than Option I. Although Option I is significantly cheaper, it does not attempt to mitigate orbital debris and would only contribute to the growing problem.

E ANALYSIS OF OPTION 3

Many future systems now include the debris prevention objectives in the initial design. One application of redesigning a spacecraft to support orbital debris reduction issues is to deorbit the spacecraft at the end of its life. In order to allow for a full design life mission and deorbit at the end, more fuel will be required. For option three, the same

satellite is modified with a larger fuel tank to accommodate the extra required fuel. No other design changes were considered for this model. The new weight will remain within the same launch vehicle's maximum load limit.

1. Decay Time: Redesign

Option three, just like option two, is a pro-active method of mitigating orbital debris. Option three will therefore establish the same deorbit guidelines as in option two. NASA's 25 year deorbit limit will be option three's decay time.

2. Fuel Requirements

With the perigee altitude established from the decay requirements, the next step is to recompute the fuel requirements. The 800 km initial orbit satellite still remains the base case. From the fuel computations in option two, 60 lbs will be required to complete a successful deorbit. In addition to the 60 lbs, some incremental fuel will be required to move the new, "heavier" weight satellite. Using equation [8], a total of 65 lbs of fuel was computed as the total incremental requirement. Computing the fuel for the 1300 km initial apogee, 160 lbs of fuel will be required to complete a successful deorbit.

3. Lifecycle Cost: Option Three

Still using the Taurus Launch Vehicle with an average cost ratio of $15,000 per kilogram, the additional 65 pounds is converted into launch costs. The cost for the additional 65 pounds is $0.442 M. This base case is designed to operate for its full design life (five years) and then deorbit. With a 10 year operational mission period and no early deorbiting, only two satellites will be required. The calculations for Option III is shown in Table 4-9. Notice that the Satellite LCC is the same as Option I. The only difference

is in the added launch weight (extra fuel). The Satellite Lifecycle costs for Option III and launch costs for two satellites over a ten year period results in a System LCC of $534.82 M.

Table 4-9. Calculations for Lifecycle Costs Using Option III.

PVSR(1) = 41.8 + 38 + 34.5 + 31.4 + 27.9 = 173.6 PVSR(2) = 26.1 + 23.6 + 21.4 + 19.5 + 17.7 = 108.3 Total PVSR = $ 281.9 M	
PVNR = 141.6 + (141.6/1.6) = $ 230.1 M	
Satellite LCC => PVSR + PVSNR = $ 512 M	
TLC = 13.6 + .44 (extra fuel) = (14.04) + (14.04/1.6) = 22.82	
System LCC for Option III => Satellite LCC + TLC = 512 + 22.82 = $ 534.82 M	
OPTION THREE SYSTEM LIFECYCLE COST	$ 534.82 M

For the 1300 km apogee example, the above Satellite LCC is the same. The only difference is in the additional fuel which is reflected in the TLC. Computing the new TLC, System LCC for a 1300 km apogee would be $ 535.87 M. This is a $ 1.05 M increase from the 800 km apogee. As a reminder, the 800 km apogee is the base case and the 1300 km case shown is for comparison only.

D. CHAPTER SUMMARY

The selection of option one, which has been established as current practice will result in the lowest total cost. However, this lower cost comes at the expense of a significant decay lifetime. This option also makes no effort towards the mitigation of debris and expecting no slow down in the launching of satellites and constellations, the debris problem will continue to grow.

Option two helps to resolve the debris problem by taking aggressive steps in deorbiting the satellite prior to the completion of its scheduled mission lifetime. Nothing comes for free! This aggressive attitude will cost about one and a half times more than a non-mitigating, natural decay option.

The last option provides the benefit of minimal cost increases when compared to Option I (current practice) while maintaining NASA's decay requirements. This option does come with a caveat. It is assumed that the fuel tank size is the only thing changed on the option three satellite.

Of these three options, Option III is clearly the best choice (see Table 4-10). With a minimal increase in cost from today's standards (option one), the decay lifetime has been significantly reduced to comply with NASA's decay limit. For the minimal cost, in the author's opinion, this is the option all space faring nations should choose.

Table 4-10. Option Comparisons.

OPTION ONE: NATURAL DECAY	**OPTION TWO:** DE-ORBIT EARLY	**OPTION THREE:** REDESIGN SATELLITE
TOTAL COST: **$ 534.1 M**	TOTAL COST: **$ 829.1 M**	TOTAL COST: **$ 534.8 M**
DECAY TIME: **574 YRS**	DECAY TIME: **25 YRS**	DECAY TIME: **25 YRS**

V. SENSITIVITY ANALYSIS

A. CRITICAL PARAMETERS

In the previous chapter, three different options regarding mitigating debris were analyzed. Each option was analyzed for cost impact and decay lifetime. Option III was clearly the best choice with only a minimal increase in cost and a significant decrease in decay lifetime. These three options from Chapter IV were based upon specific fuel, altitude, and decay critical parameters. The critical parameters were set and no further changes were made.

This chapter on the other hand is going to take the evaluation process one step further by exploring the results of changes in the different critical parameters. As each critical parameter is changed and subsequently evaluated, it will be compared with the results of Option III.

Because of the number of critical parameters and large number of possible combinations, not all combinations will be covered in this paper. These numerous combinations are a result of the common weight and cost relationship shared by each critical parameter. Instead , each critical parameter will be changed above and below the original parameters set in Chapter IV with the other critical parameters remaining constant. Evaluating each critical parameter individually will show the direct cost impact with respect to the change.

1. Relationships of Critical Parameters

When designing a spacecraft, the evaluation of these critical parameters is a must. One technique in the evaluation process is to develop graphs that represent the critical parameters in a way that allows easy transition of data from one chart to another. The following charts (Figures 5-1 thru 5-3), reveal some interesting relationships that will be of use later when making comparisons. Below, these relationships are highlighted as well as examples shown.

a. The Relationship Between Delta Velocity and Height is Linear

Examining Figure 5-1, both curves reflect a linear line. A closer look also confirms this linear relationship. Using the 800 km apogee curve, check the delta velocity at each 100 km mark. The results are an average of 0.027 km/sec for every 100 km.

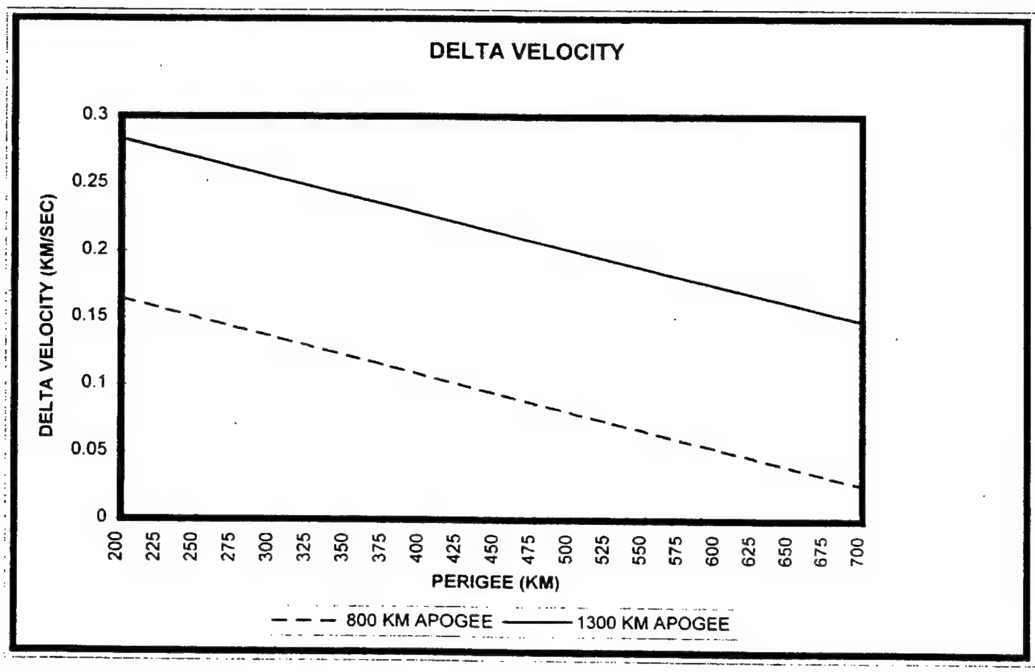

Figure 5-1. Delta Velocity Requirements.

b. Delta Velocity is Not Particularly Sensitive to Apogee

Now that the height and delta velocity relationship has been established as linear, next is to show how apogee has minimal impact on the rate. An initial look shows the curves in Figure 5-1 parallel to each other with the 1300 km curve larger by a factor of two. First bring down the results for the 800 km curve and then conduct the same check on the 1300 km curve. The results on average were the same.

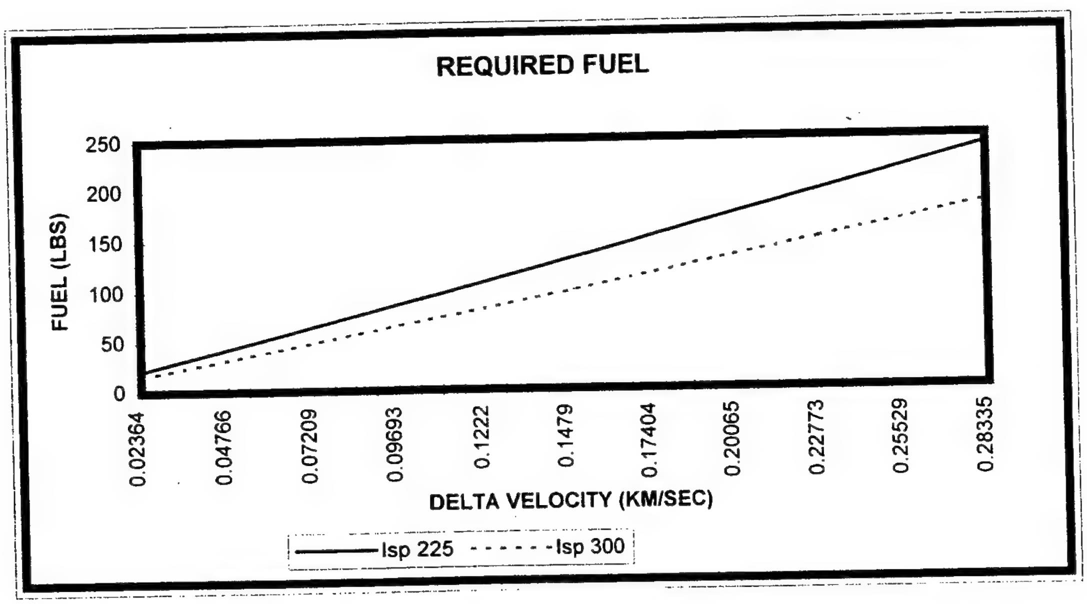

Figure 5-2. Fuel Requirements.

c. *Doubling Decay Lifetime Does Not Result in Big Changes in Delta Velocity*

For this example, enter Figure 5-3 using the 800 km curve with a 15 year initial decay time. A 15 year decay lifetime will require a perigee of 400 km which results in approximately 0.107 km/sec. Doubling the decay time to 30 years results in approximately a new perigee of 500 km. The new delta velocity is now 0.08 km/sec.

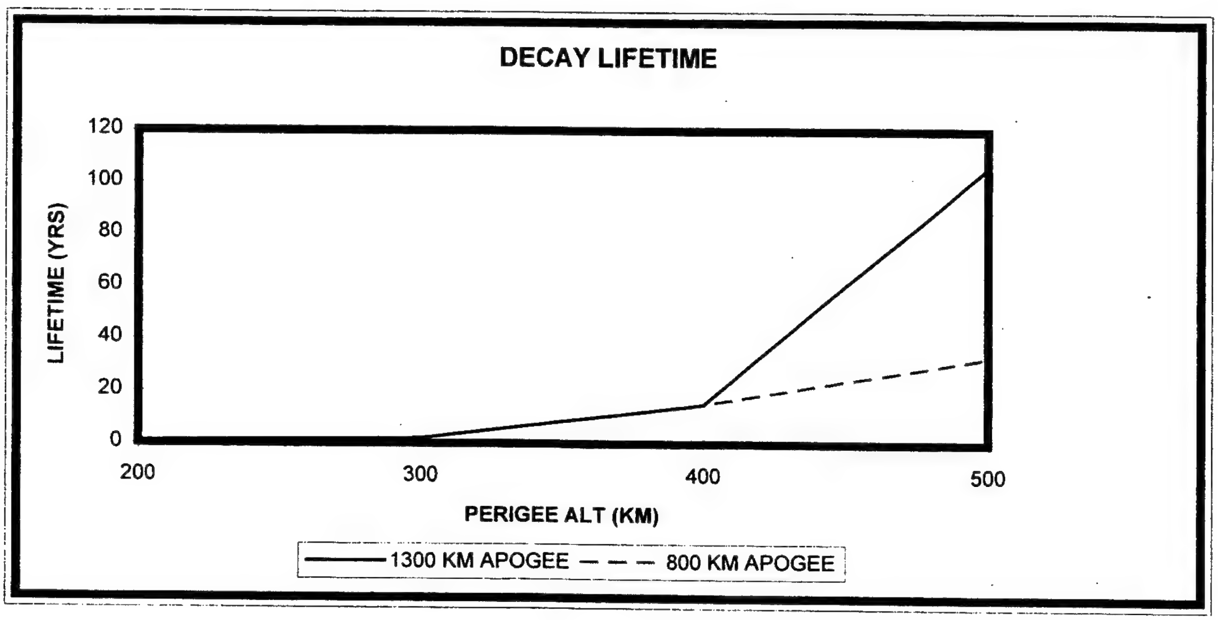

Figure 5-3. Decay Lifetime Requirements.

d. *Doubling Decay Lifetime Does Not Result in Big Fuel Changes*

To establish this statement, Figures 5-1 thru 5-3 are needed. The same values as above will be used. Start out with the initial decay lifetime of 15 year and a perigee of 400 km and move to the delta velocity graph with the perigee to obtain the

required delta velocity. This was just done above. For the initial example, the delta velocity is 0.107 km/sec. Enter Figure 5-2 with 0.107 km/sec to obtain the required fuel. For this example, use the specific impulse of 300 seconds. The result is approximately 70 lbs of fuel. Now take the new delta velocity value of 0.08 km/sec and find the new fuel requirement. The result is approximately 57 lbs of fuel. Only a 13 lb difference for doubling the decay lifetime.

The close relationship of these critical parameters is apparent and will be further highlighted in the following examples.

2. Decay

Perhaps the one critical parameter that has the most impact regarding changes is the decay lifetime of a satellite. NASA recently published a requirement for satellites to be deorbited no later than 25 years [Ref. 5]. Depending on the orbital altitude of the satellite, this restriction could be minimal or significant.

Table 5-1. Parameters for the Analysis of Changes in Decay Lifetime.

Parameter	Value
Altitude	800 km
Decay Lifetime - Increase Decay Lifetime - Decrease Decay Lifetime	 35 years 15 years
Fuel Specific Impulse	300 seconds
Fuel Weight	100 pounds

a. Analysis 1, Decay Lifetime Increase Using Option II

The baseline model (Chapter IV) showed that using Option II would result in a significant increase in cost, making it an unacceptable option. This case will explore how the effect of increasing the decay lifetime will change the cost. Using the satellite characteristics from Table 5-1, an increase in decay lifetime of 10 years under Option II would require less fuel. If the decay lifetime is raised to a 35 year limit, the perigee altitude required for a successful deorbit increases from 465 km to 525 km. This is shown in Figure 5-3. Using the same procedure that was used in the critical parameter comparison examples, all three figures will be used again. Starting with a new perigee from Figure 5-3, move to Figure 5-1 to compute the new delta velocity. Once the delta velocity is known, the new fuel requirement can be obtained from Figure 5-2. As seen from Figure 5-2, the new fuel requirement for this case is 25 lbs. With 25 pounds dedicated to deorbiting the satellite model, the remaining 75 pounds can be used for on-orbit station keeping. Converting the remaining 75 lbs into the average amount of fuel per month (1.667 lbs per month) results in 45 months of on-orbit lifetime for each satellite. Taking the total operational mission period of 120 months (10 years) and dividing it by 45 months, the result is a requirement for 2.7 satellites. Since it is more than two satellites, three will be required with an extra 15 months of performance available at the end of the period (see Table 5-2 for calculations). Because one of the thesis assumptions is a ten year operational mission period, all the different options and analysis reflect this limit. In this case, there are 15 extra months that need to be accounted for. One method is to estimate the amount of fuel needed to cover that extra

period and subtract from the last satellite as that fuel will not be needed. This method could be looked as a "credit" towards the System LCC as the extra fuel cost will be subtracted from this value. Take the 15 months and multiply it by the average amount of fuel used per month (1.667 lb/month). This results in 25 total pounds. Next convert to kilograms and then multiply by the launch cost per kilogram to get the total extra "credit". This "credit, which has a present value of \$ 0.65 M is now subtracted from the System LCC resulting a new System LCC value that reflects the adjustment back to the ten year operational mission period. Table 5-3 below shows the comparison between Analysis 1 results and the base case results from Chapter IV. Comparing the results of Table 5-2 to the Option II base case, analysis one reduces the System LCC by \$ 216 M. However when compared to Option III base case, the System LCC actually increase by \$ 25.46 M. As a result of these comparisons, Option III base case is the best choice.

Table 5-2. Calculations for Analysis 1 Lifecycle Costs.

PVSR(1) = 41.8 + 38 + 34.5 + 31.4 = 145.7 PVSR(2) = 27.9 + 26.1 + 23.6 + 21.4 = 99 PVSR(3) = 19.5 + 17.7 = 37.2 Total PVSR = \$ 281.9 M
PVNR = 141.6 + (141.6/1.5) + (141.6/2.14) = \$ 302.17 M
Satellite LCC => PVSR + PVSNR = 281.9 M+ 302.17 = \$ 584.07 M
TLC = (13.6) + (13.6/1.5) + (13.6/2.14) = 29.03
Analysis 1 System LCC => Satellite LCC + TLC = 584.07 + 29.03 = \$ 613.1 M

Table 5-3. Comparison of Decay Lifetime Increase.

Example	Option II (Deorbit Prior Msn Completion)	Option III (Redesign)
Base Case	5 Total Satellites $ 829.05 M 25 year decay lifetime	**2 Total Satellites** **$ 534.82 M** **25 year decay lifetime**
Decay Lifetime Increased Ten Years	3 Total Satellites $ 613.1 M 35 year decay lifetime	

b. Analysis 2, Decay Lifetime Increase Using Option III

Analysis 2 uses the same increase in lifetime as in Analysis 1 except applied under Option III. If decay lifetime is increased, the amount of required fuel for a successful deorbit will decrease. Since Option III sets aside the exact amount of fuel required for a successful deorbit, the on-orbit and decay time are met with only a small savings in the fuel when compared to the base case from Chapter IV. With a enough fuel available to complete the entire on-orbit period of five years, only two satellites will be required. The next step is to determine amount of required fuel for a successful deorbit. Since the decay lifetime from Analysis 1 and this case is the same, the required fuel is likewise the same. Therefore, 25 lbs will be required to successfully deorbit the satellite at the end of mission life. Now that the amount of extra fuel is computed and the number of required satellites is identified, System LCC can be computed. Table 5-4 shows the calculations for System LCC for this case.

Table 5-4. Calculations for Analysis 2 Lifecycle Costs.

PVSR(1) = 41.8 + 38 + 34.5 + 31.4 + 27.9 = 173.6 PVSR(2) = 26.1 + 23.6 + 21.4 + 19.5 + 17.7 = 108.3 Total PVSR = $ 281.9 M
PVSNR = 141.6 + (141.6/1.6) = $ 230.1 M
Satellite LCC => PVR + PVNR = 281.9 + 230.1 = $ 512 M
TLC = 13.6 + .17 (extra fuel for deorbit) = (13.77) + (13.77/1.6) = 22.38
Analysis 2 System LCC = Satellite LCC + TLC = 512 + 22.38 = $ 534.38 M

Table 5-5 shows the comparison between example two results and the base case results in Chapter IV. Extending the decay lifetime an additional ten years under Option III will result in a cost savings of $ 0.44 M.

Table 5-5. Comparison of Decay Lifetime Increase Using Option III.

Example	Option III (Redesign)
Base Case	2 Total Satellites $ 534.82 M 25 year decay lifetime
Decay Lifetime Increased Ten Years	**2 Total Satellites** **$ 534.38 M** **35 year decay lifetime**

c. *Analysis 3, Decay Lifetime Decrease Using Option II*

Imposing a stiffer requirement of a 15 year decay lifetime using Option II will increase fuel requirements for deorbit and reduce the on-orbit time for each base case. Entering Figure 5-3 with the new decay time (15 years), obtain the new required perigee. With the new perigee, move to Figure 5-1 to obtain the new delta velocity. Finally, using the new delta velocity and Figure 5-2, the new fuel requirement for

successful deorbit can be obtained. The amount of fuel required for successful decay is 80 pounds, leaving only 20 pounds of fuel for on-orbit station keeping. In computing the required number of satellites, take the fuel dedicated for on-orbit station keeping (20 lbs) and divide it by the average amount of fuel used per month (1.667 lb per month). The total on-orbit station keeping time equates to 12 months per satellite or ten total satellites for the entire operational mission period. Now that fuel requirements and number of required satellites have been identified, System LCC can be computed. Table 5-6 below shows the calculations for analysis three System LCC.

Table 5-6. Calculations for Analysis Lifecycle Cost.

PVSR(1) = 41.8 PVSR(2) = 38 PVSR(3) = 34.5 PVSR(4) = 31.4 PVSR(5) = 27.9 PVSR(6) = 26.1 PVSR(7) = 23.6 PVSR(8) = 21.4 PVSR(9) = 19.5 PVSR(10) = 17.7 Total PVSR = $ 281.9 M
PVSNR = 141.6 + (141.6/1.1) + (141.6/1.21) + (141.6/1.3) + (141.6/1.5) + (141.6/1.6) + (141.6/1.77) + (141.6/1.95) + (141.6/2.14) + (141.6/2.36) = $ 957.96 M
Satellite LCC => PVSR + PVSNR = 281.9 + 957.96 = $ 1239.86 M
TLC = (13.6) + (13.6/1.1)+(13.6/1.21)+(13.6/1.3)+(13.6/1.5)+(13.6/1.6) + (13.6/1.77)+(13.6/1.95)+(13.6/2.14)+(13.6/2.36) = 91.62
Analysis 3 System LCC => Satellite LCC + TLC = 1239.86 + 91.62 = $ 1.33 B

Table 5-7 shows the comparison between analysis three results and the base case results from Chapter IV. A reduction in decay lifetime will result in a significant increase in overall costs. Comparing the results from Table 5-6 with the Option II base case, analysis three results in a System LCC increase of $ 502 M. Option III by comparison is even more significant. The overall System LCC increase is $ 796 M. Option III base case is the best choice for this comparison.

Table 5-7. Comparison of Decay Lifetime Decrease Using Option II.

Example	Option II (Deorbit Prior Msn Completion)	Option III (Redesign)
Base Case	5 Total Satellites $ 829.05 M 25 year decay lifetime	**2 Total Satellites** **$ 534.82 M** **25 year decay lifetime**
Decay Lifetime Decreased Ten Years	10 Total Satellites $ 1.33 B 15 year decay lifetime	

d. Analysis 4, Decay Lifetime Decrease Using Option III

This case will use the same 15 year decay lifetime restriction and apply it to Option III. Under Option III, the exact deorbit fuel load requirement is added in addition to the on-orbit station keeping fuel load. With a complete full fuel load for station keeping, the satellite will be able to complete its five year mission period. Coverage for the ten year operational mission period would require two total satellites. The extra fuel that will be required for deorbit is the same as the case above, 80 lbs. With the requirements of two satellites and 80 lbs of extra fuel for each satellite, System LCC can now be computed. Table 5-8 below shows the calculations of System LCC for this case.

Table 5-8. Calculations for Analysis four Lifecycle Costs.

PVSR(1) = 41.8 + 38 + 34.5 + 31.4 + 27.9 = 173.6 PVSR(2) = 26.1 + 23.6 + 21.4 + 19.5 + 17.7 = 108.3 Total PVSR = $ 281.9 M
PVNR = 141.6 + (141.6/1.6) = $ 230.1 M
Satellite LCC => PVSR + PVSNR = 281.9 + 230.1 = $ 512 M
TLC = 13.6 + .54 (extra fuel) = (14.14) + (14.14/1.6) = 22.98
Analysis 4 System LCC => Satellite LCC + TLC = 512 + 22.98 = $ 534.98 M

Table 5-9 shows the comparisons between example four results and the base case results from Chapter IV. Decreasing the decay lifetime by 10 years will increase system LCC by $ 0.16 M, a minimal impact on cost. The base case Option III remains the better choice considering the minimal differences in cost and the fact that the decay lifetime meets NASA guidelines.

Table 5-9. Comparison of Decay Lifetime Decrease Using Option III.

Example	Option III (Redesign)
Base Case	**2 Total Satellites** **$ 534.82 M** **25 year decay lifetime**
Decay Lifetime Decreased Ten Years	2 Total Satellites $ 534.98 M 15 year decay lifetime

3. Fuel

Unlike the other critical parameters, fuel may be the most controllable parameter from a designer's perspective. If a fuel load is changed in either direction, the satellite's performance characteristics will change. In addition to the change in fuel weight, the

change in specific impulse will also change performance. For example, if fuel weight is reduced, the original altitude will have to be lowered to achieve the original on-orbit performance or the specific impulse could be increased to help offset the reduction in fuel weight.

a. Analysis 5, Fuel Increase Using Option II

Using Option II (Deorbit Satellite Prior to Mission Completion), the addition of an extra 20 pounds will help towards the on-orbit period. As stated above, the base case from Chapter IV will be used. Table 5-10 shows the parameters for the four examples regarding fuel load changes (five thru eight).

Table 5-10. Parameters for the Analysis of Changes Fuel Load.

Parameter	Value
Altitude (Apogee)	800 km
Decay Lifetime (NASA Guidelines)	25 years
Fuel Specific Impulse	300 seconds
Fuel Load - Increase in Fuel Weight - Decrease in Fuel Weight	 120 pounds 80 pounds

Using equations [7] and [8], 60 pounds of fuel will be needed to successfully deorbit the satellite model with a maximum decay lifetime of 25 years. With the total fuel load of 120 pounds and a requirement for 60 pounds to deorbit, 60 pounds is left for on-orbit station keeping. Take the on-orbit fuel (60 lbs) and divide it by the average amount of fuel required per month for station keeping (1.667 lbs/month). Total available on-orbit time equals 36 months. Taking the total operational mission period (120

months) and dividing it by the total on-orbit time for the satellite results a requirement of 3.3 satellites. Rounding up results in the actual requirement of four satellites with 18 months of on-orbit capability left after the ten year period. Again this extra time will be result in a "credit" that will be subtracted from the System LCC. The additional fuel weight is negligible when computing the total satellite costs, however it does add to the total launch costs. Now that the fuel requirements and number of satellites are established, System LCC can be computed for this case. Table 5-11 below shows the System LCC calculations for this case.

Table 5-11. Calculations for Analysis 5 Lifecycle Costs.

PVSR(1) = 41.8 + 38 + 34.5 = 114.3 PVSR(2) = 31.4 + 27.9 + 26.1 = 85.4 PVSR(3) = 23.6 + 21.4 + 19.5 = 64.5 PVSR(4) = 17.7 Total PVSR = \$ 281.9 M
PVSNR = 141.6 + (141.6/1.3) + (141.6/1.77) + (141.6/2.36) = \$ 390.5
Satellite LCC => PVSR + PVSNR = 281.9 + 390.5 = \$ 672.4 M
TLC = 13.6 + .12 (extra fuel) = (13.72) + (13.72/1.3)+(13.72/1.77) + (13.72/2.36) = \$ 37.83 M
Analysis 5 System LCC => Satellite LCC + TLC = 672.4 + 37.83 = \$ 710.23 M

Once the System LCC has been computed, the "credit" needs to be subtracted to determine the new System LCC that reflects the full ten year mission period. Just as was done in a previous case, take the extra time (18 months) and multiply it with the average amount of fuel required per month (1.667 lb/month). This results in a total of 30 pounds. Convert this to kilograms and determine the launch cost per kilogram. The result is a present value of \$ 0.077 M. Now take the System LCC and subtract this new result from

it. This is the new System LCC ($ 710.15 M) that has been adjusted to the ten year period. Table 5-12 below compares analysis five results with the base case results from Chapter IV. The results of analysis five is a System LCC savings of $119 M when compared to Option II base case and a System LCC increase of $ 175 M when compared to Option III base case. Option III base is the best choice in these comparisons.

Table 5-12. Comparison of Fuel Increase Using Option II.

Example	Option II (Deorbit Prior Msn Completion)	Option III (Redesign)
Base Case	5 Total Satellites $ 829.05 M 25 year decay lifetime	**2 Total Satellites** **$ 534.82 M** **25 year decay lifetime**
Fuel Increase of 20 Pounds	4 Total Satellites $ 710.15 M 25 year decay lifetime	

b. Analysis 6, Fuel Increase Using Option III

As a review, Option III is a redesign of the satellite model by way of a larger fuel load to allow for a full on-orbit period and a successful deorbit. This results in 60 pounds of fuel set aside for deorbit and 120 pounds of fuel to station keep over a period of time that only requires 100 pounds. Since the extra 20 pounds of fuel is not enough to extend the satellite over the entire ten year operational period, two satellites will have to be used. This means that there is a surplus of fuel that could be used to increase the decay lifetime. This decision is made because of the ten year mission limit. For this case, the 60 pounds plus the extra 20 pounds of fuel will be the additional requirement. Now that the fuel and satellite requirements have been identified, LCC can be computed. Table 5-13 below shows the calculations for analysis six System LCC.

Table 5-13. Calculations for Analysis 6 Lifecycle Costs.

PVSR(1) = 41.8 + 38 + 34.5 + 31.4 + 27.9 = 173.6 PVSR(2) = 26.1 + 23.6 + 21.4 + 19.5 + 17.7 = 108.3 Total PVSR = $ 281.9 M
PVSNR = 141.6 + (141.6/1.6) = $ 230.1 M
Satellite LCC => PVSR + PVSNR = 281.9 + 230.1 = $ 512 M
TLC = 13.6 + .54 (extra fuel) = (14.14) + (14.14/1.6) = 22.98
Analysis 6 System LCC => Satellite LCC + TLC = 512 + 22.98 = $ 534.98 M

Table 5-14 below compares analysis six results with the base case results from Chapter IV. A fuel increase under Option III will increase System LCC by $ 0.16, a minimal cost increase.

Table 5-14. Comparison of Fuel Increase Using Option III.

Example	Option III (Redesign)
Base Case	**2 Total Satellites** **$ 534.82 M** **25 year decay lifetime**
Fuel Increase of 20 pounds	2 Total Satellites $ 534.98 M 25 year decay lifetime

c. *Analysis 7, Fuel Decrease Using Option II*

If the satellite model fuel load was reduced by 20 pounds under Option II, the available fuel for on-orbit station keeping and deorbit would be only 80 pounds. Deorbit fuel requirements have been established at 60 pounds. This leaves only 20 pounds for on-orbit station keeping. As in analysis five, computing the average pounds per month ratio results in 12 months for the available 20 pounds of fuel. Over the period

of ten years, ten total satellites will be required. In this case there will not be any extra fuel, instead 20 pounds of weight will be subtracted from the original satellite weight. Just as before, once the fuel requirements are known and the number of satellites determined, System LCC can be computed. Table 5-15 below shows the calculations for analysis seven System LCC.

Table 5-15. Calculations for Analysis 7 Lifecycle Costs.

PVSR(1) = 41.8 PVSR(2) = 38 PVSR(3) = 34.5 PVSR(4) = 31.4 PVSR(5) = 27.9 PVSR(6) = 26.1 PVSR(7) = 23.6 PVSR(8) = 21.4 PVSR(9) = 19.5 PVSR(10) = 17.7 Total PVSR = $ 281.9 M
PVSNR = 141.6 + (141.6/1.1) + (141.6/1.21) + (141.6/1.3) + (141.6/1.5) + (141.6/1.6) + (141.6/1.77) + (141.6/1.95) + (141.6/2.14) + (141.6/2.36) = $ 957.92 M
Satellite LCC => PVSR + PVSNR = 281.9 + 957.92 = $ 1239.82 M
TLC = 13.6 - .14 (fuel subtracted) = (13.46) + (13.46/1.1) + (13.46/1.21) + (13.46/1.3) + (13.46/1.5) + (13.46/1.6) + (13.46/1.77) + (13.46/1.95) + (13.46/2.14) + (13.46/2.36) = 91.04
Analysis 7 System LCC => Satellite LCC + TLC = 1239.82 + 91.04 = $ 1.33 B

Table 5-16 below compares analysis seven results with the base case results in Chapter IV. Any reduction in fuel is going to impact overall costs, especially under Option II where extra is not provided for deorbit. Under Option II, the System LCC

increase of analysis seven over the base case is $ 502 M. Likewise the comparison of analysis seven results to the Option III base case results in an even larger System LCC increase which totals $ 796 M.

Table 5-16. Comparison of Fuel Decrease Using Option II.

Example	Option II (Deorbit Prior Msn Completion)	Option III (Redesign)
Base Case	5 Total Satellites $ 829.05 M 25 year decay lifetime	**2 Total Satellites** **$ 534.82 M** **25 year decay lifetime**
Fuel Decrease of 20 Pounds	10 Total Satellites $ 133 B 25 year decay lifetime	

d. *Analysis 8, Fuel Decrease Using Option III*

Under Option III, fuel for deorbit is set aside to ensure decay lifetime requirements are met. Therefore decreasing the fuel load under Option III will directly impact only the available on-orbit station keeping time. Reducing the fuel load by 20 pounds results in 80 pounds available for station keeping. Dividing the 80 pounds into the average amount of fuel required per month (1.667 lb/month), the result is a satellite that can remain on-orbit for 48 months. Over the ten year operational period, three satellites will be required. Because the actual coverage is only two and a half satellites, extra fuel be reflected as a "credit" and will have to be subtracted from the System LCC. With the fuel and number of satellites identified, System LCC can be calculated. Table 5-17 below show calculations for analysis eight System LCC.

Table 5-17. Calculations for Analysis 8 Lifecycle Costs.

PVSR(1) = 41.8 + 38 + 34.5 + 31.4 = 145.7 PVSR(2) = 27.9 + 26.1 + 23.6 + 21.4 = 99 PVSR(3) = 19.5 + 17.7 = 37.2 Total PVSR = $ 281.9 M
PVSNR = 141.6 + (141.6/1.5) + (141.6/2.14) = $ 302.17 M
Satellite LCC => PVSR + PVSNR = 281.9 + 302.17 = $ 584.07 M
TLC = 13.6 + .41(deorbit fuel) - .14 (decrease in fuel load) = (13.87) + (13.87/1.5) + (13.87/2.14) + (13.87/2.14) = 29.6
Analysis 8 System LCC => Satellite LCC + TLC = 584.07 + 29.6 = $ 613.67 M

The next step is to adjust the System LCC value by subtracting the "credit" of extra fuel available at the end of the ten year operational mission period. Taking that extra half period for the satellite (24 months), multiply it with the average amount of fuel required per month (1.667 lb/month) for a result of 40 pounds. Convert the 40 pounds into kilograms and multiply it with the launch cost per kilogram ($ 15000 /kg). Take the final present value result $ 0.10 M and subtract it from the System LCC value. This results in a new System LCC of $ 613.57 M and is shown in Table 5-18 below. Any reduction in fuel will ultimately result in increased costs as shown by a $ 78 M increase for analysis eight over the Option III base case. As a result, the Option III base case is the better choice of the two.

Table 5-18. Comparison of Fuel Decrease Using Option III.

Example	Option III (Redesign)
Base Case	**2 Total Satellites** **$ 534.82 M** **25 year decay lifetime**
Fuel Decrease of 20 pounds	3 Total Satellites $ 613.57 M 25 year decay lifetime

4. Altitude

Altitude may be very restrictive depending on the satellite payload requirements. However, if the payload requirements allow for some flexibility in altitude, moving the satellite to a lower altitude would reduce the amount of required fuel for a successful deorbit.

Delta velocity on average changes 0.05 km/sec for every 200 km change in altitude. Table 5-19 below lists the parameters used in the examples regarding changes in altitude (examples nine thru twelve).

Table 5-19. Parameters for the Analysis of Changes in Altitude.

Parameter	Value
Altitude - Increase Altitude - Decrease Altitude	 900 km 700 km
Decay Lifetime	25 years (NASA Guideline)
Fuel Specific Impulse	300 seconds
Fuel Weight	100 pounds

a. Analysis 9, Altitude Increase Using Option II

An increase in altitude of 100 km will require a larger fuel load in order to successfully deorbit the satellite model within the 25 year decay limit. Although Figures 5-1 thru 5-3 are not used, the procedure is the same as before. The first step is to determine the new perigee of 460 km, converting the perigee to a delta velocity, and then finally to a fuel amount. This new fuel requirement is 77 pounds. Option II does not allow for any extra fuel for deorbit which results in using the fuel reserved for on-orbit station keeping. With a deorbit requirement of 77 pounds, this leaves only 23 pounds for on-orbit station keeping. Dividing the on-orbit fuel by the average amount of fuel per month results in a satellite on-orbit period of 14 months. Covering the ten year operational period will result in a requirement for nine total satellites. With fuel and satellite quantity issues identified, System LCC can be computed. Table 5-20 below shows the calculations for analysis nine System LCC.

The amount of extra time available for this case was negligible when compared to the System LCC. For this reason, it is not addressed in this case. Table 5-21 below compares analysis nine's results with the base case results in Chapter IV. As shown in Table 5-21 below, an altitude increase will increase the overall cost. Specifically, under Option II, analysis nine results in increases of System LCC by $ 410 M over the Option II base case. Comparing these same results to Option III base case, an increase of $ 705 M is seen for analysis nine. Option III base case is the best choice in this comparison.

Table 5-20. Calculations for Analysis 9 Lifecycle Costs.

PVSR(1) = 41.8 PVSR(2) = 38 PVSR(3) = 34.5 PVSR(4) = 31.4 PVSR(5) = 27.9 PVSR(6) = 26.1 PVSR(7) = 23.6 PVSR(8) = 21.4 PVSR(9) = 19.5 PVSR(10) = 17.7 Total PVR = $ 281.9
PVSNR = 141.6 + (141.6/1.1) + (141.6/1.21) + (141.6/1.3) + (141.6/1.5) + (141.6/1.6) + (141.6/1.95) + (141.6/2.14) + (141.6/2.36) = $ 877.92 M
Satellite LCC => PVSR + PVSNR = 281.9 + 877.92 = $ 1159.82 M
TLC = (13.6) + (13.6/1.1) + (13.6/1.21) + (13.6/1.3) + (13.6/1.5) + (13.6/1.6) + (13.6/1.95) + (13.6/2.14) + (13.6/2.36) = 84.32
Analysis 9 System LCC => Satellite LCC + TLC = 1159.82 + 84.32 = $ 1.24 B

Table 5-21. Comparison of Altitude Increase Using Option II.

Example	Option II (Deorbit Prior Msn Completion)	Option III (Redesign)
Base Case	5 Total Satellites $ 829.05 M 25 year decay lifetime	**2 Total Satellites** **$ 534.82 M** **25 year decay lifetime**
Altitude Increase of 100 km	9 Total Satellites $ 1.24 B 25 year decay lifetime	

b. Analysis 10, Altitude Increase Using Option III

With an altitude increase of 100 km under Option III, no initial change from the base case is noted. Under Option III, extra fuel is allocated above and beyond the on-orbit fuel load to ensure successful decay. Increasing the altitude 100 km will

increase the fuel requirements for decay. Since this case is using the same altitude as analysis nine, the required fuel of 77 pounds for deorbit can be used in this case. Therefore this case will have an addition of 77 pounds to the satellite. Also, because this case is under Option III, the satellite will be able to complete the entire five year design life. This will require two satellites for the ten year period. With fuel and satellite number issues identified, System LCC is the next to be computed. Table 5-22 below shows the calculations for System LCC.

Table 5-22. Calculations for Analysis 10 Lifecycle Costs.

PVSR(1) = 41.8 + 38 + 34.5 + 31.4 + 27.9 = 173.6 PVSR(2) = 26.1 + 23.6 + 21.4 + 19.5 + 17.7 = 108.3 Total PVR = $ 281.9 M
PVNR = 141.6 + (141.6/1.6) = $ 230.1 M
Satellite LCC => PVSR + PVSNR = 281.9 + 230.1 = $ 512 M
TLC = 13.6 + .54 (extra fuel) = (14.14) + (14.14/1.6) = 22.98
Analysis 10 System LCC => Satellite LCC + TLC = 512 + 22.98 = $ 534.98 M

Table 5-23 below shows the comparison of this case and the Option III base case. Increasing altitude under Option III will increase costs minimally as shown by a $ 0.16 M System LCC increase over Option III base case.

Table 5-23. Comparison of Altitude Increase Using Option III.

Example	Option III (Redesign)
Base Case	**2 Total Satellites** **$ 534.82 M** **25 year decay lifetime**
Altitude Increase of 100 km	2 Total Satellites $ 534.98 M 25 year decay lifetime

c. *Analysis 11, Altitude Decrease Using Option II*

A decrease in altitude by 100 km will reduce deorbit fuel requirements for a successful deorbit. Under Option II, this reduction in fuel load requirements allows for more fuel on-orbit. Similar to analysis eight, Figures 4-1 thru 4-3 can not be used, however the procedure is exactly the same. After determining the new perigee and delta velocity as was done previously, the new fuel load for deorbit is 34 pounds. This leaves 66 pounds for station keeping. Next take the 66 pounds and divide it by the average amount of fuel per month (1.667 lb/month). This results in 40 months of on-orbit coverage for the model, totaling three satellites required for the 120 month (ten year) operational period. The next step is to compute the System LCC which is shown below in Table 5-24.

Table 5-24. Calculations for Analysis 11 Lifecycle Costs.

PVSR(1) = 41.8 + 38 + 34.5 + 31.4 = 145.7 PVSR(2) = 27.9 + 26.1 + 23.6 + 21.4 = 99 PVSR(3) = 19.5 + 17.7 = 37.2 Total PVSR = $ 281.9 M
PVSNR = 141.6 + (141.6/1.5) + (141.6/2.14) = $ 302.17 M
Satellite LCC => PVSR + PVSNR = 281.9 + 302.17 = $ 584.07 M
TLC = (13.6) + (13.6/1.5) + (13.6/2.14) = 29.03
Analysis 11 System LCC => Satellite LCC + TLC = 584.07 + 29.03 = $ 613.1 M

Table 5-25 below compares analysis eleven's results with the base case results. A decrease in altitude will result in an System LCC savings of $ 216 M under Option II but will still exceed Option III base case by $ 78 M. Option III base case is the best choice in this comparison.

Table 5-25. Comparison of Altitude Decrease Using Option II.

Example	Option II (Deorbit prior msn completion)	Option III (Redesign)
Base Case	5 Total Satellites $ 829.05 M 25 year decay lifetime	**2 Total Satellites** **$ 534.82 M** **25 year decay lifetime**
Altitude Decrease of 100 km	4 Total Satellites $ 613.1 M 25 year decay lifetime.	

d. Analysis 12, Altitude Decrease Using Option III

Decreasing altitude by 100 km under Option III will provide only minimal changes overall. On-orbit fuel will remain the same and the requirement for deorbit fuel remain at 34 pounds, the same as the case above. Again, with the complete load of on-orbit fuel available for station keeping, the satellite will complete its five year design life. Therefore, for the ten year operational mission period, two satellites will be required. Next is the System LCC calculations which are shown in Table 5-26 below.

Table 5-27 below shows the comparison between analysis twelve results and the base case results in Chapter IV. As mentioned in analysis eleven, operating at a lower altitude will result in overall cost savings. Decreasing the altitude under Option III will result in a System LCC savings of $ 0.35 M. In this case, Analysis 12 is the better choice over Option III base case.

Table 5-26. Calculations for Analysis 12 Lifecycle Costs.

PVSR(1) = 41.8 + 38 + 34.5 + 31.4 + 27.9 = 173.6 PVSR(2) = 26.1 + 23.6 + 21.4 + 19.5 + 17.7 = 108.3 Total PVSR = $ 281.9 M
PVSNR = 141.6 + (141.6/1.6) = $ 230.1 M
Satellite LCC => PVSR + PVSNR = 281.9 + 230.1 = $ 512 M
TLC = 13.6 + .23 = (13.83) + (13.83/1.6) = 22.47
Analysis 12 System LCC => Satellite LCC + TLC = 512 + 22.47 = $ 534.47 M

Table 5-27. Comparison of Altitude Decrease Using Option III.

Example	Option III (Redesign)
Base Case	2 Total Satellites $ 534.82 M 25 year decay lifetime
Altitude Decrease of 100 km	**2 Total Satellites** **$ 534.47 M** **25 year decay lifetime**

B. CHAPTER SUMMARY

This chapter is a sensitivity analysis of the results from Chapter IV. This analysis involved identifying three critical parameters that have a significant impact on satellite cost and operations. The base case from Chapter IV was brought forward and the critical parameters identified in this chapter were changed and compared to the results from Chapter IV.

Twelve different analysis were developed and compared with the base case in Chapter IV. Between the analysis and two different options, in the author's opinion, Option III was the best choice in every case (see Table 5-28). Of the Option III results,

ten analysis showed the base case as the best choice. Overall, minimal change occurred in the results for Option III as compared to significant changes in Option II.

These results clearly show that it is far cheaper to build-in debris mitigating practices from the design level than to mitigate with existing spacecraft as in Option II which requires the acquisition incrementally, additional satellites.

Table 5-28. Critical Parameter Summary.

Example	Option II (Deorbit Prior Msn Completion)	Option III (Redesign)
Base Case	5 Total Satellites $ 829.05 M 25 year decay lifetime	**2 Total Satellites** **$ 534.82 M** **25 year decay lifetime**
Decay Lifetime Increase (+ 10 years)	3 Total Satellites $ 613.1 M 35 year decay lifetime	2 Total Satellites $ 534.38 M 35 year decay lifetime
Decay Lifetime Increase (- 10 years)	10 Total Satellites $ 1.33 B 15 year decay lifetime	2 Total Satellites $ 534.98 M 15 year decay lifetime
Fuel Increase (+ 20 lbs)	4 Total Satellites $ 710.15 M 25 year decay lifetime	2 Total Satellites $ 534.98 M 25 year decay lifetime
Fuel Decrease (- 20 lbs)	10 Total Satellites $ 1.33 B 25 year decay lifetime	3 Total Satellites $ 613.57 M 25 year decay lifetime
Altitude Increase (+ 100 km)	9 Total satellites $ 1.24 B 25 year decay lifetime	2 Total Satellites $ 534.98 M 25 year decay lifetime
Altitude Decrease (- 100 km)	4 Total Satellites $ 613.1 M 25 year decay lifetime	2 Total Satellites $ 534.47 M 25 year decay lifetime

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VI. SUMMARY AND CONCLUSIONS

A. SUMMARY

As can be seen throughout this paper, setting a policy regarding the mitigation of orbital debris will result in a cost impact. This cost impact represents one of the larger issues when considering policy. However, there are other issues that are equally important, but are beyond the scope of this paper. Those issues are briefly discussed below.

When policy is set, the issue of enforcement arises. The questions of how and who would conduct the policing actions are not easy ones to answer. Taking the analogy of a police officer in a city, laws have been set and yet people seem to break them, including the "smaller" violations like parking or speeding. It is reasonable then to expect that not everyone operating in space will comply with the policy 100%. Deviations to the policy may be very small, but they will still be deviations. In the author's opinion, most likely those deviations will be done as a cost cutting measure. Since space is considered a common area to be shared and used by all, the use and abuse of it is a difficult one to manage or police and the incentives are similar to the "commons problem" in England.

With the use of space expected to increase, the probability of a collision likewise increases. This raises the issues of liability. If a collision occurs, proof of ownership and of who struck who is going to be extremely difficult considering the collision will most likely be from a small unidentifiable piece of debris.

The final issue that will be addressed is determining the start time of a debris mitigating policy. Specifically, the issue is setting policy immediately or at some future date. Setting policy immediately without a full and complete understanding of the true orbital debris problem may require unnecessary restrictions resulting in higher costs. Setting policy at some future date would provide more time for data collection and prediction refinements resulting in a more specific policy that would not be as restrictive or broad scoped. On the other hand, setting a policy soon is important since it will only impact spacecraft not in design. This in the author's opinion is a very worthwhile trade off given the uncertainty in accurate debris estimation.

B. CONCLUSIONS

Throughout this paper, several issues with respect to setting a debris policy have been addressed or mentioned. Of those issues, cost impact is one of the most important issues and is the primary focus of this paper.

The final cost impact after developing a satellite model and subsequent validation of the model's critical parameters was Option III. This option is a "redesign" of the satellite model by way of a larger fuel load to allow for a full on-orbit period and a successful deorbit within NASA's 25 year limit. In every case examined, Option III was clearly the better choice. What made Option III the better choice was the minimal System LCC increase of only $ 0.71M while still mitigating orbital debris within NASA's 25 year decay limits. Of significance is the minimal increases in cost for Option III from Option I. Option I is represented as the current method of operating in space.

Planning in most cases is better than retrofitting. Option II is retrofitting, Option III is planning. Incorporating mitigation practices (deorbit) into design will have minimal cost impact. Incorporating mitigation practices after the design will have significant impact and in some cases may not be able to be accomplished. The conclusion is that mitigating orbital debris can be cost effective as shown throughout this thesis. incorporating deorbit into the design has minimal cost impact.

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APPENDIX A. ORBITAL COMPUTATIONS

Table A-1. Delta Velocity and Fuel Requirements With Respect to Changes In Perigee.

Apogee	Perigee	essentricity	ΔV	I_{sp} 225 sec	I_{sp} 300 sec
800 km	700 km	0.007	0.0261 km/s	23.51 lbs	17.66 lbs
800 km	600 km	0.014	0.0523 km/s	46.83 lbs	35.23 lbs
800 km	500 km	0.020	0.0749 km/s	66.73 lbs	50.26 lbs
800 km	400 km	0.029	0.1088 km/s	96.19 lbs	72.59 lbs
800 km	300 km	0.036	0.1354 km/s	119 lbs	89.93 lbs
800 km	200 km	0.044	0.1658 km/s	144.73 lbs	109.56 lbs

Table A-2. Delta Velocity and Fuel Requirements With Respect To Changes In Perigee.

Apogee	Perigee	essentricity	ΔV	I_{sp} 225 sec	I_{sp} 300 sec
1300 km	1200 km	0.007	0.0253 km/s	22.79 lbs	17.12 lbs
1300 km	1100 km	0.013	0.0470 km/s	42.14 lbs	31.69 lbs
1300 km	1000 km	0.020	0.0724 km/s	64.54 lbs	48.60 lbs
1300 km	900 km	0.028	0.1016 km/s	89.97 lbs	67.87 lbs
1300 km	800 km	0.034	0.1235 km/s	108.83 lbs	82.19 lbs
1300 km	700 km	0.041	0.1492 km/s	130.72 lbs	98.86 lbs
1300 km	600 km	0.048	0.1750 km/s	152.44 lbs	115.46 lbs
1300 km	500 km	0.055	0.2009 km/s	174 lbs	131.97 lbs
1300 km	400 km	0.062	0.2269 km/s	195.38 lbs	148.40 lbs
1300 km	300 km	0.070	0.2567 km/s	219.58 lbs	167.06 lbs
1300 km	200 km	0.077	0.2829 km/s	240.59 lbs	183.30 lbs

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APPENDIX B. CALCULATIONS FOR RECURRING COSTS

[All Equations from Ref. 10]

A. STRUCTURE

Spacecraft Structure	300 lbs

$Y = (5.838)(X1)$ [9]

Where $X1$ = Structure Weight

Y = CER value for Spacecraft Structure

Therefore $Y = 1756.1$

B. THERMAL

Thermal Weight	
- Active Thermal Weight	13.1 lbs
- Passive Thermal Weight	16.9 lbs
- Total Thermal Weight	30 lbs

$Y = 76.171 + (12.187)(X1) + (4.511)(X2)$ [10]

Where $X1$ = Active Weight

$X2$ = Passive Weight

Y = CER value for Thermal Suite

Therefore $Y = 312.1$

C. ADCS

ADCS	
- Determination Suite Weight	45 lbs
- RCS Suite Weight	30 lbs
- Total ADCS Weight	75 lbs

$Y = (250.542)(X1^{0.735})$ [11]

Where $X1$ = Attitude Determination Suite Weight

Y = CER value for ADCS (Attitude Determination)

Therefore $Y = 4111.4$

$Y = (27.667)(X1^{0.619})(X2^{0.473})$

Where $X1$ = Reaction Control System Suite Weight

$X2$ = Design Life

Y = CER value for ADCS(Reaction Control)

Therefore $Y =$

D. ELECTRICAL POWER SYSTEM

EPS *- Number of Solar Cells* *- Generation Suite Weight*	3000 231 lbs

$Y = (7.894)(X1^{0.588})$ [12]

Where $X1$ = Number of Solar Cells

Y = CER value for Power Generation

Therefore $Y = 874.7$

EPS *- Beginning of Life Power* *- Storage Suite Weight*	1200 W 135 lbs

$Y = (2.722)(X1^{0.848})$ [13]

Where $X1$ = Beginning of Life Power

Y = CER value for Power Storage

Therefore $Y = 1111.8$

EPS Suite Weight	265 lbs

$Y = (58.755)(X1^{0.713})$ [14]

Where $X1$ = PCD Suite Weight

Y = CER for Power Conditioning and Distribution (PCD)

Therefore $Y = 31392$

E. TELEMETRY, TRACKING AND CONTROL

TT&C Transmitter (2) *- UHF* *- SHF*	 2.1 lbs 3.1 lbs

$$Y = 76.928 + (20.435)(X1) \quad [15]$$

Where $X1$ = Transmitter Weight

Y = CER value for a TT&C Transmitter

Therefore $Y = 119.8$ (UHF)

Therefore $Y = 140.3$ (S-Band)

TT&C Receiver/Exciter	6.6 lbs

$$Y = (47.359)(X1^{1.105})(X2^{0.420}) \quad [16]$$

Where $X1$ = Receiver/Exciter Suite Weight

Y = CER value for a TT&C Receiver/Exciter

Therefore $Y = 509.8$

TT&C Transponder (2)	6.2 lbs

$$Y = (377.529)(X1^{0.281}) \quad [17]$$

Where $X1$ = Transponder Weight

Y = CER value for TT&C Transponder

Therefore $Y = 630.4$

TT&C Digital Electronics *- Suite Weight* *- Number of Digital Electronic Boxes* *- Number of Links*	 20.7 lbs 5 2

$$Y = (23.406)(X1^{0.922})(X2^{0.659})(X3^{1.091}) \qquad [18]$$

Where $X1$ = Digital Electronics Suite Weight

$X2$ = Number of Digital Electronic Boxes

$X3$ = Number of Links

Y = CER value for TT&C Digital Electronics

Therefore $Y = 2353.1$

TT&C Analog Electronics *- Suite Weight* *- Solenoid Driver (4)* *- Squib Driver (4)*	 0.5 lbs 1.4 lbs 1.4 lbs

$$Y = (113.777)(X1^{0.519}) \qquad [19]$$

Where $X1$ = Analog Electronics Weight

Y = CER value for TT&C Analog Electronics

Solenoid Driver (qty 2)

$Y1 = (Y)(qty^{0.926})$

Squib Driver (qty 2)

$Y = (13.777)(X2^{0.519})$

Where $X2$ = Squib Driver Weight

Solenoid = 286.6, Squib = 489.1

TT&C Antenna (Horn & Radiator) *- Horn & Radiator* *- Gain* *- Wavelength* *- Effective Area*	 2.0 lbs 0.3 db/10 .5 FT .5 SQFT

$$Y = (119.351)(X1^{0.708})(X2^{0.240}) \quad [20]$$

Where X1 = Antenna Weight

X2 = Effective Area

Y = CER value for TT&C Antenna (Horn & Radiator)

Therefore Y = 842.7

TT&C Antenna (Dipoles)	.83 lbs

$$Y = (26.609)(X1^{1.070}) \quad [21]$$

Where X1 = Antenna Dipoles Weight

Y = CER value for TT&C Antenna (Dipoles)

Therefore Y = 21.8

TT&C Antenna (S-Band) *- S-Band Weight* *- Gain* *- Wavelength* *- Effective Area*	 .5 lbs 0.26 db/10 .5 FT .45 SQFT

$$Y = (64.560)(X1^{1.009})(X2^{0.315}) \quad [22]$$

Where X1 = Antenna Weight

X2 = Effective Area

Y = CER value for TT&C Antenna (S-band)

Therefore Y = 24.9

TT&C RF Distribution	1.0 lb

$$Y = (-7.386) + (29.180)(X1) + (70.676)(X2) \qquad [23]$$

Where X1 = RF Distribution Weight

X2 = Active (1= yes, 0 = No)

Y = CER value for TT&C RF Distribution

Therefore Y = 92.5

F. COMMUNICATIONS

Communications Transmitter (TWTA) *- TWTA Weight* *- Output Power* *- Frequency* *- WPF*	 11.0 lb 25 W 2.15 Ghz 23

$$Y = (22.196)(X1^{0.727})(X2^{0.280}) \qquad [24]$$

Where X1 = TWTA Weight

X2 = Weighted Composite Variable

Y = CER value for Communications Transmitter (TWTA)

Therefore Y = 305.3

Communications Transmitter (Solid State) *- Solid State Transmitter Weight* *- Output Power* *- Component Quantity*	 50 lbs 35 lbs 1

$$Y = (338.550) + (25.557)(X1^{9.985})(X2) \qquad [25]$$

Where X1 = Solid State Transmitter Weight

X2 = Output Power

Y = CER value for Communications Transmitter (Solid State)

Therefore Y = 1965.9

Communications Receiver/Exciter Weight	13.1 lbs

$Y = (193.30)(X1^{0.675})$ [26]

Where X1 = Receiver/Exciter Suite Weight

Y = CER value for Communications Receiver/Exciter

Therefore Y = 1097.5

Communications Transponder Weight (2)	30 lb 21.5 lb

$Y = (67.433)(X1)$ [27]

Where X1 = Transponder Weight

Y = CER value for Communications Transponder

Therefore Y = 2023.0

Communications Digital Electronics Weight	11.3 lb

$Y = (515.079)(X1^{0.379})$ [28]

Where X1 = Digital Electronics Suite Weight

Y = CER value for Communications Digital Electronics

Therefore Y = 1291.2

Communications *- Weight of Other Antenna Components* *- Weight of Horn, Dish* *- Antenna Suite Weight*	 14 lbs 24 lbs 38 lbs

$Y = (35.473)(X1) + (24.835)(X2)$ [29]

Where X1 = Weight of Other Antenna Components

X2 = Weight of Horn, UHF dish

Y = CER value for Communications Antenna

Therefore $Y = 1092.7$

Communications Antenna (Reflectors) *- Antenna Reflector Diameter Squared*	8.0 SQFT

$$Y = (75.849)(X1^{0.935}) \qquad [30]$$

Where $X1$ = Antenna Reflector Diameter Squared

Y = CER value for Communications Antenna Reflectors

Therefore $Y = 530.1$

Communications RF Distribution *- RF Distribution Suite Active Weight* *- RF Distribution Suite Wave Guide Weight*	 3.0 lbs 3.0 lbs

$$Y = (82.601)(X1) + (11.856)(X2) \qquad [31]$$

Where $X1$ = RF Distribution Suite Active Weight

$X2$ = RF Distribution Suite Wave Guide Weight

Y = CER value for Communications RF Distribution

Therefore $Y = 283.4$

G. APOGEE KICK MOTOR (AKM)

AKM Total Weight	440.6 lbs

$$Y = (2.355)(X1^{0.820}) \qquad [32]$$

Where $X1$ = AKM Total Weight

Y = CER value for Apogee Kick Motor (AKM)

Therefore $Y = 346.8$

H. INTEGRATION ASSEMBLY AND TEST (IA&T)

IA&T	
- Spacecraft Weight	1508 lbs
- Communications Total Weight	192 lbs
- Weight	1700 lbs

Y = (4.833)(X1) [33]

Where X1 = Spacecraft Weight + Communications Total Weight

Y = CER value for Integration Assembly and Test (IA&T)

Therefore Y = 8216.1

I. PROGRAM LEVEL

Spacecraft Vehicle Total Recurring Cost	

Y = (0.289)(X1) [34]

Where X1 = Space Vehicle Total Recurring Cost

Y = CER value for Program Level

Therefore Y =

J. LOOS - (3 - AXIS STABILIZED SATELLITES)

Space Vehicle Weight	1700 lbs

Y = (2.212)(X1) [35]

Where X1 = Spacecraft Weight + Communication Total Weight

Y = CER value for Operations and Orbital Support

Therefore Y = 3760.4

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APPENDIX C. CALCULATIONS FOR NONRECURRING COSTS

[All Equations from Ref. 11]

A. STRUCTURE

Spacecraft Structure	300 lbs

$$Y = (99.045)(X1)^{0.789} \quad [36]$$

Where $X1$ = Structure Weight

Y = CER value for Spacecraft Structure

Therefore $Y = 8918.28$

B. THERMAL

Thermal Weight	30 lbs

$$Y = (0.243)(X1)^{0.597}(X2)^{0.983} \quad [37]$$

Where $X1$ = Thermal Weight

$X2$ = Space Vehicle Weight

Y = CER value for Thermal Subsystem

Therefore $Y = 2773.22$

C. ADCS

ADCS	
- Determination Suite Weight	45 lbs
- Reaction Control System	30 lbs
- Total ADCS Weight	75 lbs

$$Y = (666.439)(X1)^{0.711} \quad [38]$$

Where $X1$ = Determination Suite Weight

Y = CER value for Attitude Determination Suite

Therefore $Y = 9981.47$

$$Y = (125.998)(X1)^{0.733} \quad [39]$$

Where $X1$ = Reaction Control System Weight

Y = CER value for ADCS (Reaction Control System)

Therefore $Y = 1524.38$

D. ELECTRICAL POWER SYSTEM

EPS	
- Number of Solar Cells	3000
- Generation Suite Weight	231 lbs
- Beginning of Life Power	1200 W
- Storage Suite Weight	135 lbs
- EPS Suite Weight	265 lbs

$$Y = (0.025)(X1) + (0.024)(X2) \quad [40]$$

Where $X1$ = (Generation Suite Weight)(Beginning of Life Power)

$X2$ = Number of Solar Cells

Y = CER value for Electrical Power Generation

Therefore $Y = 7002$

$$Y = 114.127 + (2.584)(X1) \quad [41]$$

Where $X1$ = (Weight of One Battery)(Capacity of One Battery)

Y = CER value for Electrical Power Storage

Therefore $Y = 1276.93$

$$Y = (5.515)(X1) \quad [42]$$

Where $X1$ = Beginning of Life Power

Y = CER value for Power Conditioning and Distribution

Therefore $Y = 6618$

E. TELEMETRY, TRACKING AND CONTROL

TT&C	
- Transmitter	5.2 lbs
- Receiver/Exciter	6.6 lbs
- Digital Electronics (2 Links)	20.7 lbs
- Antenna (4 systems)	2.0 lbs

$$Y = (67.121)(X1) \quad [43]$$

Where X1 = Transmitter Suite Weight

Y = CER value for TT&C Transmitter

Therefore Y = 349.03

$$Y = (-224.351) + (116.683)(X1) \quad [44]$$

Where X1 = Receiver/Exciter Suite Weight

Y = CER value for TT&C Receiver/Exciter

Therefore Y = 545.76

$$Y = (211.243)(X1)^{0.787}(X2)^{0.853} \quad [45]$$

Where X1 = Digital Electronics Suite Weight

X2 = Number of Links

Y = CER value for TT&C Digital Electronics

Therefore Y = 4142.19

$$Y = (-222.262) + (30.670)(X1) + (480.840)(X2) \quad [46]$$

Where X1 = Antenna Suite Weight

X2 = Number of Antenna Systems

Y = CER value for TT&C Antenna

Therefore Y = 2314.5

F. COMMUNICATIONS

Communications System	
- TWTA	11 lbs
- Solid State Transmitter	50 lbs
- Receiver/Exciter	13.1 lbs
- Transponder (2 units)	30 lbs
- Digital Electronics (10 links)	11.3 lbs
- Antenna (4 systems)	38 lbs
- Antenna Reflectors	8.0 SQFT

$$Y = (524.161)(X1)^{0.875} \qquad [47]$$

Where $X1$ = TWTA Weight

Y = CER value for Communications Transmitter (TWTA)

Therefore $Y = 4272.51$

$$Y = (0.249)(X1)^{1.101}(X2)^{0.728} \qquad [48]$$

Where $X1$ = Solid State Transmitter Weight

$X2$ = Transmitter Frequency

Y = CER value for Communications Transmitter (Solid State)

Therefore $Y = 4845.53$

$$Y = (273.793)(X1) \qquad [49]$$

Where $X1$ = Receiver/Exciter Suite Weight

Y = CER value for Communications Receiver/Exciter

Therefore $Y = 3586.69$

$$Y = (682.769)(X1)^{0.463} \qquad [50]$$

Where $X1$ = Transponder Weight

Y = CER value for Communications Transponder

Therefore $Y = 3297.47$

$$Y = (211.243)(X1)^{0.787}(X2)^{0.853} \quad [51]$$

Where $X1$ = Digital Electronics Suite Weight

$X2$ = Number of Links

Y = CER value for Communications Digital Electronics

Therefore $Y = 10152.08$

$$Y = (-222.262) + (30.670)(X1) + (480.840)(X2) \quad [52]$$

Where $X1$ = Antenna Suite Weight

$X2$ = Number of Antenna Systems

Y = CER value for Communications Antenna

Therefore $Y = 2866.56$

$$Y = (1763.889)(X1) \quad [53]$$

Where $X1$ = Antenna Reflector Diameter Squared

Y = CER value for Communications Antenna Reflector

Therefore $Y = 14111.11$

G. INTEGRATION ASSEMBLY AND TEST (IA&T)

IA&T	
- Spacecraft Weight	1508 lbs
- Communications Total Weight	192 lbs
- Total Weight	1700 lbs

$$Y = (956.384) + (0.191)(X1) \quad [54]$$

Where $X1$ = Spacecraft + Communications Total Nonrecurring Cost

Y = CER value for Integration Assembly and Test (IA&T)

Therefore $Y = 17874.73$

H. PROGRAM LEVEL

Spacecraft Vehicle Total Recurring Cost	$ 36993

$$Y = (2.340)(X1)^{0.808} \quad [55]$$

Where X1 = Space Vehicle Total Nonrecurring Cost

Y = CER value for Program Level

Therefore Y = 23262.37

I. AEROSPACE GROUND EQUIPMENT (AGE)

Space Vehicle Total Nonrecurring Cost	$88577.71

$$Y = (8.304)(X1)^{0.638} \quad [56]$$

Where X1 = Space Vehicle Total Nonrecurring Cost

Y = CER value for Aerospace Ground Equipment

Therefore Y = 11903.66

APPENDIX D. DEFINITION OF TERMS

Apogee - The point in the orbit that is the farthest from the center of the Earth. The apogee altitude is the distance of the apogee point above the surface of the Earth.

Critical Density - A critical population density is reached when that population will produce fragments from random collisions at a rate that is increasing and is greater than the removal rate by natural processes.

Debris Flux - The number of impacts per square meter per year expected on a randomly oriented planar surface of an orbiting space structure.

Delta Velocity - The change in the velocity vector caused by thrust measured in units of meters per second.

Eccentricity - The apogee altitude minus perigee altitude of an orbit divided by twice the semimajor axis. Eccentricity is zero for circular orbits and less than one for all elliptical orbits.

Geosynchronous Earth orbit (GEO) - An orbit with a period equal to the sidereal day. A circular GEO orbit with zero degrees inclination is a geostationary orbit, i.e., the nadir point is fixed on the Earth's surface. The altitude of a circular GEO orbit is 35,788 km. When GEO is referred to as an altitude, it is that of circular GEO orbit.

Inclination - The angle the orbit plane makes with the equatorial plane.

Low Earth orbit (LEO) - The region of space below the altitude of 2000 km.

Meteoroids - Naturally occurring particulates associated with solar system formation or evolution processes. Meteoroid material is associated with asteroid breakup or material released from comets.

Orbital debris -Man made particulates released in orbit.

Perigee - The point in the orbit that is nearest to the center of the Earth. The perigee altitude is the distance of the perigee point above the surface of the Earth.

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